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THE INSTRUMENTATION OF THE S-2
AIRCRAFT FOR STABILITY AND CONTROL
FLIGHT TESTING

by

Thomas John Duncan

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Thomas John Duncan

April 1969

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THE INSTRUMENTATION OF THE S-2 AIRCRAFT

FOR STABILITY AND CONTROL FLIGHT TESTING

by

Thomas John Duncan
Lieutenant, United States Navy
B.S. Ed., East Central State, 1961

Submitted in partial fulfillment of the
requirements for the degree of

MASTER OF SCIENCE IN AERONAUTICAL ENGINEERING

from the

NAVAL POSTGRADUATE SCHOOL
April 1969

NPS ARCHIVE
1969
DUNCAN, T.

~~Thesis D7892~~ c.1

ABSTRACT

A Navy US-2A aircraft was instrumented for use in stability and control flight testing. Various methods of recording and sensing the aerodynamic parameters necessary to evaluate stability and control flight testing were investigated. An Ampex Series 800 Magnetic Tape Recorder, obtained on a loan basis, was calibrated and installed to provide a means of recording airborne data. Using available equipment, sensing devices were installed in the aircraft and calibrated to measure control forces, control surface position and normal acceleration. The completed installation provides the means for sensing and recording those aerodynamic parameters most difficult to measure without electronic aids. It also allows for the incorporation of additional sensing and recording devices should they become available.

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ACKNOWLEDGEMENTS

The author wishes to thank Professor Donald M. Layton, Associate Professor of Aeronautics at the Naval Postgraduate School, for his guidance and support as thesis advisor. The author would also like to express his appreciation to those technicians of the Aeronautics Department most closely associated with this project, particularly Mr. Burtis H. Funk and Mr. Dana R. Mayberry. In addition the author is deeply indebted to Mr. John D. Russell of the Flight Test Instrumentation Division of the Naval Missile Center and Mr. William E. Carpenter of the Flight Test Division of the NASA Ames Research Center for their professional advice and assistance.

I. INTRODUCTION

At the Naval Postgraduate School the Aeronautics Department, as a part of the flight mechanics curricula, offers a sequence of courses in Flight Evaluation Techniques. These courses provide a theoretical background to the quantitative and qualitative techniques for evaluation of aircraft performance, static and dynamic stability, and control characteristics in flight. An integral part of this program is the Flight Evaluation Technique Laboratory, which involves flying Naval Aircraft for the purpose of obtaining performance and stability data. The aircraft used in this program are those assigned to the Naval Auxiliary Landing Field, Monterey, California for Combat Readiness Training of Naval Postgraduate School Aviators, and as such, have no special instrumentation installed for the sensing and recording of aerodynamic data other than the standard cockpit flight and engine instruments. In order to provide the students enrolled in the above mentioned courses with a reasonable background in the current methods of data collection and interpretation, it is necessary to provide an aircraft suitably instrumented for this educational task.

In addition, the Aeronautics Department is engaged in various research projects which require the use of an instrumented aircraft. Such projects presently include:

1. The analysis of an aircraft's dynamic response to turbulence.
2. An examination of the "Military Specifications for Flying Qualities of Piloted Airplanes, MIL-F-8785(ASG)" in relationship to the various pilot transfer functions that have been developed by varying levels of pilot experience and training.

Permission was granted to the Naval Postgraduate School by the Commander, Naval Air Force, U. S. Pacific Fleet to instrument one

US-2A aircraft. The aircraft assigned to the Aeronautics Department was BUNO. 136533 and is shown in Figure 1.

To provide an airborne data acquisition system that could be utilized for both the Flight Evaluation Techniques Laboratory and for research projects, various methods of data recording and sensing had to be examined. Requirements for a system that (1) could be procured at little cost, (2) could be installed and calibrated using available personnel and facilities, (3) could be operated without unduely complex preflight procedures, and (4) would provide a reasonable degree of reliability and accuracy had to be met.

The initial phase in providing such an aircraft was the installation and calibration of electronic devices that could be utilized in stability and control flight testing. This phase was accomplished at the Naval Postgraduate School, Monterey, California during the period June 1968 through March 1969.

II. AIRCRAFT DESCRIPTION

The US-2A aircraft showin in Figure 1 is a four place, twin-engine, high wing monoplane manufactured by the Gruman Aircraft Engineering Corporation of Bethpage, New York. The first prototype of this aircraft, the XS-2A, was flown in December of 1952. Since that time the Gruman Aircraft Engineering Corporation has produced over one thousand of these aircraft in various configurations. The S-2A was originally designed as a carrier based aircraft for use in anti-submarine warfare. The US-2A aircraft is a utility version of the S-2A used primarily for pilot training and administrative support. All anti-submarine warfare equipment has been removed. The overall dimensions of the aircraft are shown in Figure 2.

The aircraft is powered by two Wright R1820-82A nine-cylinder reciprocating engines, capable of producing 1525 brake horsepower at an engine speed of 2800 revolutions per minute and a manifold pressure of 56.5 inches of mercury at sea level on a standard day. Each engine is equipped with a three-bladed, variable pitch, full feathering propeller. Engine instruments are located on the pilot's and co-pilot's instrument panel. These instruments include:

1. A dual manifold pressure indicator marked in increments of inches of mercury.
2. A dual tachometer indicating engine speed in revolutions per minute.
3. Dual fuel and oil pressure indicators in increments of pounds per square inch.
4. Dual oil, cylinder head, and carburetor air temperature indicators in increments of degrees centigrade.

The aircraft's electrical power supply systems consist of a 24-volt direct current, monitored bus system and a 115-volt, fixed frequency, alternating current system. The dc system is powered by two 300 ampere, engine-driven generators regulated to 27.7 volts and by a 24 volt/36 ampere hour storage battery. The two dc generators are connected in parallel and direct current is distributed through a system of busses. The ac system consists of a main, a standby, and a pilot's instrument inverter. The inverters are 115 volt, three phase, 400 Hertz motor-generator combinations that operate on direct current from the dc bus system. Instruments to monitor the electrical power supply system include:

1. Two combined voltmeter and ammeter indicators which indicate the voltage and dc output of their respective generators.
2. An ac voltmeter which indicates the voltage on each phase of the main inverter.

The aircraft's flight control system consists of mechanical, hydraulic, and electrically actuated control surfaces and is shown in Figure 3 and Figure 4.

The longitudinal control system consists of conventional elevators which are mechanically actuated by the pilot's or co-pilot's control wheel through a system of push rods, bellcranks, and cables. An electrically actuated trim tab and a geared tab are installed on both the right and left elevator.

The lateral control system consists of ailerons and two spoilers in each wing outer panel. The ailerons are mechanically actuated by the pilot's or co-pilot's control wheel through a push rod and bellcrank linkage. The spoilers are mechanically linked with the ailerons. As each aileron travels upward the adjacent pair of spoilers move upward with it extending through the upper surface of the wing. When the aileron travels downward the adjacent pair of spoilers remain flush with the upper surface of the wing. An electrically actuated trim tab is installed on the left aileron. Spring tabs are installed on both ailerons.

The directional control system consists of a conventional rudder which is actuated by the pilot's or co-pilot's rudder pedals through a system of push rods and cables. A rudder trimmer which can be actuated either electrically or hydraulically is located just forward of the rudder. In normal flight the rudder trimmer is deflected electrically by means of a trim switch and provides directional trimming. In conditions where greater directional control is required, such as during single-engine operation, the rudder trimmer is deflected hydraulically so that its position is proportional to the rudder deflection. Under these conditions the aircraft is thus provided with an increased effective rudder area. Either mode of operation must be selected electrically. A balance tab located on the rudder is slaved to the rudder trimmer.

Three trim tab position indicators located in the cockpit show the setting of the elevator tabs, the aileron tab and the rudder trimmer in degrees of tab or trimmer movement.

Flight instruments are located on the pilot's or co-pilots instrument panel and include:

1. Two airspeed indicators calibrated in knots.
2. Two altimeters calibrated to indicate the aircraft's pressure altitude in feet.
3. Two rate of climb indicators calibrated in feet per minute.
4. Two turn and slip indicators, where the turn needle indicates motion about the aircraft's vertical axis and the ball indicates the aircraft's lateral acceleration.
5. The pilot's vertical gyro and the co-pilot's gyro horizon which indicate the aircraft's pitch and bank attitude in degrees.
6. Two direction indicators which show the aircraft's magnetic heading in degrees.
7. An accelerometer which indicates the acceleration in g's along the aircraft's vertical axis.

Other instruments located on either the pilot's or co-pilot's instrument panel include:

1. An outside air temperature indicator calibrated in degrees centigrade.

2. An elapsed time clock with a sweep second hand.

3. Two fuel quantity gages which indicate pounds of fuel remaining in both fuel tanks.

III. DESIRED INSTRUMENTATION

The instrumented aircraft required by the Aeronautics Department had to be capable of being utilized for a dual purpose. First, it was to be used as a "flying classroom" with which the techniques for the flight evaluation of an aircraft's performance and stability characteristics could be examined. Second, it was to be used as a flight test vehicle for current and future research projects in flight mechanics.

The Flight Evaluation Techniques courses and laboratory sequences have two objectives. The primary purpose is to familiarize the students with the methods of flight evaluation. Thus the instrumentation accuracy required for the sensing and recording of engine and aircraft parameters is that which will provide the proper responses and trends. The absolute correctness of the data is not emphasized. The secondary purpose is to familiarize the students with some of the current methods of data collection and interpretation. In this regard the equipment used to sense and record the engine and aircraft parameters should be of the type currently used in flight testing.

Current research projects need a data acquisition system that is both modern and accurate.

To satisfy this dual requirement it was necessary to provide a means of continuous data recording which was reliable, accurate, flexible, and yet possessed growth potential. A recording device possessing these qualities could be used to: (1) record those parameters investigated in performance and stability flight testing that are not presented on the aircraft's normal flight instrument panel, (2) provide a current method of data collection and analysis, and (3) provide the recording accuracy necessary for research projects.

The recording device best suited to meet these requirements was either a recording oscillograph or a magnetic tape recorder.

To determine which parameters were to be sensed and recorded an examination was made of the current and projected performance and stability flight tests that were being conducted in the Flight Evaluation Techniques Laboratory sequence. It was concluded that measurements of the following parameters are needed: (1) engine speed, (2) engine manifold pressure, (3) carburetor air temperature, (4) carburetor deck pressure, (5) fuel flow, (6) fuel remaining, (7) outside air temperature, (8) airspeed, (9) altitude, (10) vertical speed, (11) control forces, (12) primary and secondary control surface position, (13) normal acceleration, (14) angle of attack, (15) angle of sideslip, (16) pitch, roll, and yaw rates.

An examination of the parameters to be measured by the various research projects was not made due to their differing requirements, but it is felt that most of these requirements will necessitate only minor variations in the installation.

The initial phase in providing a data acquisition system for the US-2A aircraft was to: (1) install and calibrate the desired recording device, and (2) install and calibrate sensing devices for stability and control flight testing that would measure those parameters not presented on the aircraft's flight instrument panel and those most difficult to measure accurately without electronic aids. A comparison of the parameters visually presented to the pilot or co-pilot in the cockpit with those required for stability and control flight testing, and an examination of the accuracy afforded by these visual presentations, revealed a need to measure control forces, primary surface control position, normal acceleration, angle of attack, angle of sideslip, pitch rate, roll rate, and yaw rate. Additional sensing devices were to be added to the basic system as they became available.

IV. ACTUAL INSTRUMENTATION

To procure the desired equipment with which to instrument the US-2A aircraft, various government facilities on the West Coast that were involved in flight testing were contacted. From the Flight Test Instrumentation Division of the Naval Missile Center, a seven channel Ampex Series 800 Magnetic Tape Recorder was obtained on a loan basis. In addition, transducers were obtained to sense rudder force and elevator position. From the Flight Test Division of the NASA Ames Research Center, transducers were obtained to sense control wheel forces, aileron position, rudder position, angle of attack, and angle of sideslip. Transducers with which to sense normal acceleration were available at the Naval Postgraduate School. Devices to measure pitch, roll, and yaw rates, as well as the necessary supplemental equipment to install and calibrate an angle of attack and sideslip probe were not available locally and could not be obtained on a loan basis. The limited finances that were available for this project precluded their purchase. Supplementary equipment was available to complete the installation and calibration of both the magnetic tape recorder and the associated electrical devices necessary to measure control forces, control surface position and normal acceleration.

A. DATA RECORDING

The tape recorder that was installed in the aircraft was an Ampex Series 800 Magnetic Tape Recorder. It is shown in Figure 5 and consists of the following components:

1. A shockmounted tape transport mechanism which consists of a closed loop tape drive system and a plug-in type head assembly. The tape transport is shown in Figure 6.

2. A shockmounted cabinet designed to accomodate seven plug-in record amplifiers, either of the frequency modulation or amplitude/pulse width modulation type in any desired combination. The cabinet with its associated amplifiers is shown in Figure 7.

3. A shockmounted electronics power supply which furnishes regulated voltages for the record amplifiers and the tape transport record head. The electronics power supply is shown in Figure 8.

4. A shockmounted capstan power unit which consists of a 400 Hertz power supply, a power amplifier, and a 60 Hertz oscillator which are used to supply frequency stabilized 60 Hertz power to the capstan motor in the tape transport. The capstan power supply is shown in Figure 9.

5. A remote control unit which controls the distribution of power to all components in the system by means of a "power" switch, and which controls tape transport operation and the amplifier record relays by means of a "record" switch. The remote control unit is shown in Figure 10.

6. A test unit which provides a pre-operational checkout of the recorder by systematically examining selected system voltages, the carrier head current in the FM amplifiers, the bias level of the oscillator in the electronics power supply, and signal head current of the AM record amplifiers. The test unit is shown in Figure 11.

The tape recorder's primary power requirements of 28 volt dc and 115 volt, 400 Hertz are provided by the aircraft's dc bus system and main inverter, respectively, through a power terminal board. This regulated power is then distributed to the various components of the tape recorder through a system of plug-in type interconnecting cables. The cabling system is shown in Figure 12. Circuit protection is provided by a five amp. dc circuit breaker marked RCDR located on the co-pilot's circuit breaker panel as indicated in Figure 13, and a five amp. fuse marked RCDR located on the ac distribution box as indicated in Figure 14.

The presently installed system utilizes a ½ inch tape operating at a speed of fifteen inches per second. With a 10½ inch reel of acetate base tape this will provide thirty two minutes of recording time while a one mil thick Mylar base tape will provide forty eight minutes of recording time. Six FM record amplifiers and one AM record amplifier are installed in the record amplifier cabinet thus providing six channels for high frequency aerodynamic data and one channel for wide band direct voice recording. The six FM amplifiers are calibrated to provide a center carrier frequency of 13,500 Hertz with a corresponding tape speed of fifteen inches per second. The frequency is varied by the amplitude of the dc voltage input signal. A $\pm$ 40 percent deviation of the center carrier frequency at the maximum signal inputs of $\mp$ 1.4 volts dc provides a large frequency range with a corresponding high signal to noise ratio. The AM amplifier is calibrated to provide a frequency response of $\pm$ 3 decibels at 300 to 15,000 Hertz with a corresponding tape speed of fifteen inches per second.

The electrical signals that are generated by the various transducers used to sense aerodynamic data are modified so that they provide a dc voltage amplitude within the range $\pm$ 1.4 volts. The electrical signals are then transmitted to a signal terminal board. From the signal terminal board the input signals are transmitted via co-axial cables to connectors located at the rear of the record amplifier cabinet. The signal terminal board and associated co-axial cables provide for thirteen signals. The cables may be connected to any of the six FM record amplifiers thus providing flexibility in which aerodynamic parameters are to be recorded.

Voice transmissions are obtained from the pilot's intercommunication system microphone and transmitted to the AM record amplifier via the signal terminal board and corresponding co-axial cable.

Upon receipt of the Ampex Series 800 Magnetic Tape Recorder a system checkout was conducted utilizing the test unit. All voltages, head currents, and the oscillator bias level were found to be in the proper range. Utilizing an electronic counter, the center carrier frequency of each FM amplifier was adjusted to 13,500 Hertz by means of the FREQ ADJ control on the front panel of the amplifier. With the counter still connected to the amplifier a positive 1.4 volt dc input signal was connected to the amplifier and the carrier frequency was checked at minus forty percent deviation. Using the RECORD LEVER control on the front panel of the amplifier, the frequency was adjusted to 8100 $\pm$ 81 Hertz. The center frequency was then rechecked. The process was repeated on each amplifier until both the center frequency and the minus forty percent deviation frequency were in the proper range.

Upon completion of the system checkout, signal inputs of thirty second duration in increments of 0.1 volts dc within the range $\pm$ 1.4 volts dc were connected to each of the FM amplifiers and recorded on the magnetic tape. The tape was then played back on the ground based Ampex Series FR1100 Magnetic Tape Record/Reproduce unit shown in Figure 15. The signal outputs for each of the six FM amplifiers were displayed on an electronic digital voltmeter and are listed in Table I to provide a comparison with the known signal input. Table I indicates that the accuracy obtained utilizing the Ampex Series 800 Magnetic Tape Recorder and the Ampex Series

TABLE I

TAPE RECORDER CHANNEL OUTPUT

COMPARED WITH INPUTS

Input signal Volts dc (positive)	Output signals Volts dc (positive) Channel number* 2	3	4	5	6	7
0.100	0.108	0.088	0.097	0.102	0.097	0.090
.200	.210	.194	.200	.205	.201	.196
.300	.313	.302	.304	.315	.307	.303
.400	.416	.406	.405	.420	.410	.406
.500	.511	.506	.503	.520	.508	.503
.600	.608	.605	.601	.620	.606	.601
.700	.705	.704	.698	.716	.702	.696
.800	.798	.800	.790	.812	.797	.789
.900	.893	.897	.886	.909	.893	.883
1.000	.990	.992	.978	1.004	.988	.978
1.100	1.083	1.087	1.072	1.100	1.082	1.071
1.200	1.178	1.187	1.168	1.196	1.178	1.165
1.300	1.272	1.282	1.260	1.290	1.272	1.256
1.400	1.363	1.378	1.350	1.386	1.368	1.349

*Seven channel tape recorder. Channel 1 is AM voice recording. Channels 2-7 are FM recording.

TABLE I (continued)

Input signal Volts dc (negative)	Output signals		Volts dc (negative) Channel number			
	2	3	4	5	6	7
0.100	0.106	0.135	0.112	0.112	0.113	0.122
.200	.211	.243	.214	.216	.215	.227
.300	.314	.349	.314	.322	.317	.331
.400	.416	.456	.415	.430	.420	.437
.500	.520	.562	.518	.533	.522	.540
.600	.623	.667	.619	.640	.621	.642
.700	.727	.773	.718	.740	.721	.743
.800	.828	.878	.816	.841	.822	.846
.900	.929	.987	.912	.942	.920	.949
1.000	1.030	1.092	1.010	1.048	1.019	1.051
1.100	1.132	1.194	1.108	1.143	1.115	1.152
1.200	1.233	1.296	1.203	1.247	1.212	1.252
1.300	1.333	1.398	1.299	1.348	1.308	1.353
1.400	1.429	1.500	1.393	1.443	1.402	1.450

FR1100 playback equipment, while not perfect, is quite good, particularly in the range from a minus 0.4 volts to a plus 0.9 volts dc. The accuracy generally decreases as the maximum carrier frequency deviation of $\pm$ 40 percent is approached. This is to be expected as the allowable frequency range provides for a $\pm$ 1 percent error at the maximum deviation. Another source of error is the frequency drift of the center carrier frequency. This frequency drift may be as great as 1.5 percent and still remain within the published accuracy of the recording equipment.

The signal output of Channel Four was displayed graphically by using an Electronic Associates Incorporated 1110 Variplotter with a scale factor of 0.2 volts per inch and a time factor of 0.05 inche per second. This graphical output, shown in Figure 16, illustrates the feasibility of providing aerodynamic data on a time dependent basis utilizing locally available equipment. In addition it illustrates a noise level of approximately 0.020 volts which tends to mask the system's accuracy.

Voice signals were transmitted via the pilot's intercommunication system microphone to the AM amplifier and recorded. The magnetic tape was then played back through the Ampex Series FR 1100 playback unit which was connected to a speaker. The quality of the playback was acceptable with little distortion.

B. DATA SENSING

Upon completion of the installation and calibration of the data recording system, devices to measure control forces, control surface position, and normal acceleration were installed and calibrated.

1. Control Forces

To measure elevator and aileron control wheel forces a control wheel force transmitter was procured from the Flight Test Division of the NASA Ames Research Center. The control wheel, shown in Figure 17, employs two cantilever beams which are deflected fore or aft as force is applied to move the elevator control surfaces and which are deflected up or down as force is applied to move the aileron control surfaces. Eight 120 ohm resistance strain gages are bonded to the cantilever beams forming two Wheatstone bridges which provide voltage amplitude signals that are related to the forces applied. The internal mechanisms of the control wheel force transmitter are shown in Figure 18.

To measure rudder pedal forces four 120 ohm resistance strain gages were bonded to the co-pilot's rudder pushrods in the form of a Wheatstone bridge to provide a voltage amplitude signal which is related to the difference in forces applied to the co-pilot's rudder pedals. This installation is shown in Figure 19.

The range of control wheel and rudder pedal force levels to be measured on the US-2A aircraft were determined from "Military Specifications for Flying Qualities of Piloted Airplanes, MIL-F-8785(ASG)". These specifications indicate the necessity of measuring longitudinal control forces of up to twenty pounds of push and fifty pounds of pull, maximum lateral control forces of fifty pounds and maximum directional control forces of 180 pounds. To provide a measurement of the force levels required within the signal range of the magnetic tape recorder, ± 1.4 volts dc, it was necessary to amplify the strain gage signals by using Statham Model CAO-3-12594 Strain Gage Signal Amplifiers. These amplifiers use a 28 volt dc power source and were designed to provide signal outputs from zero to five volts dc. By employing the adjustable balance and gain features of these amplifiers it was possible to obtain output signals from a minus 0.600 volts to a positive 1.400 volts dc. This range of output signals, while satisfactory for the measurement of longitudinal control forces, was not satisfactory for the measurement of lateral and directional control forces.

The elevator control wheel forces to be measured varied from twenty pounds of push to fifty pounds of pull. The push forces were related to a negative signal output while the pull forces were related to a positive signal output. The amplifier gain was then adjusted so that the signal output would remain within the range of a minus 0.600 volts to a positive 1.400 volts dc when the maximum force levels were applied.

The aileron control wheel forces to be measured varied from zero to fifty pounds in either a clockwise or counter-clockwise direction. To utilize the full range of ± 1.4 volts dc to measure aileron control wheel forces in either direction, a switch to change the polarity of the output signal was placed in the electrical circuit between the strain gage bridge and the amplifier. By the proper positioning of this switch either a clockwise or a counter-clockwise force as applied to the control wheel could be related to a positive signal output. The amplifier gain was then adjusted so that the maximum aileron control wheel force of fifty pounds resulted in a signal output of approximately 1.4 volts dc.

The rudder pedal forces to be measured varied from zero to 180 pounds on either the right or left pedal. The right rudder pedal was related to a positive signal output while the left rudder pedal was related to a negative signal output. However the strain levels were so low on the rudder pedal pushrods that application of the maximum rudder pedal forces resulted in a signal output of less than one volt dc while utilizing the maximum amplifier gain. This limited signal outputs to those related to force levels of up to 150 pounds on the left rudder pedal and up to 180 pounds on the right rudder pedal. To facilitate the measuring of rudder pedal forces of up to 180 pounds on either rudder pedal a switch was placed in the electric circuit between the strain gage bridge and the amplifier. Activation of this switch provides a polarity change in the output signal so that control forces on either rudder pedal can be related to a positive signal output.

A schematic of the electrical circuit used to measure aileron control wheel forces and rudder pedal forces is shown in Figure 20. The electrical circuit to measure elevator control wheel forces is identical except for the absence of the output signal switch.

To provide calibration curves that would relate a given control force to a signal output it was necessary to apply forces of known magnitude to the control wheel force transmitter and the co-pilot's rudder pedals and record the resulting signal output. Calibration curves for control wheel forces were determined by using the electrical circuit shown in Figure 20 while the control wheel was mounted in a vertical position. It was possible to apply forces of known magnitudes in either the clockwise or counterclockwise direction by adding calibrated weights to loading trays on a pulley and cable system. The resulting output signal was displayed on an electronic digital voltmeter. The calibration set-up is shown in Figure 21. The calibration curves plotted in Figure 22 indicate that the curves for both the clockwise and counterclockwise applied forces have the same slope and are linear. The control wheel curve for the elevator was determined from the same system, except that the control wheel force transmitter was mounted in a horizontal position. The loading trays were attached so that either push or pull forces of a known magnitude could be applied as shown in Figure 23. The resulting calibration curve, shown in Figure 24, has a constant slope and indicates a linear relationship between the applied forces and the signal output. Upon completion of the calibration curves, the control wheel force transmitter with its supplemental electrical devices was installed at the co-pilot's station of the US-2A aircraft as shown in Figure 25.

Calibration curves for rudder pedal forces were determined by applying forces of known magnitudes to each rudder pedal and reading the electrical output of the strain gage bridge mounted on the co-pilot's rudder pushrods through the electrical circuit shown in Figure 20. The aircraft's gust lock system was engaged to lock the rudder control surface in the neutral position. Utilizing a dial push-pull gage calibrated up to fifty pounds, known forces were applied to each of the co-pilot's rudder pedals and the resulting output signals were recorded from an electronic digital voltmeter connected to the signal terminal board. This calibration curve, plotted in Figure 26, shows the linear relationship between the rudder pedal forces and the signal outputs. It also shows that the slope for the right pedal and the left pedal are not equal. Since the electrical signal generated by the Wheatstone bridge is dependent on the strain experienced by the right pushrod in one case and by the left pushrod in the other case, any variation in strain per force between the two rods would show up as a difference in slope. This assumes that the strain gage installation was done properly so as to cancel the effects of bending.

2. Control Surface Position

To measure control surface position, transducers were connected to either a pushrod or a control cable that has a linear displacement in respect to the angular position of the control surface. The placement of these transducers is indicated in Figure 3 and Figure 4. A 10,000 ohm resistance potentiometer of the sliding arm type was attached to the pushrod located between the elevator control sector and the bellcrank assembly to determine the angular position of the elevators. This installation is shown in Figure 27. The angular position of the ailerons and rudder are determined by transducers of the type shown in Figure 28. The internal mechanisms of these transducers, shown in Figure 29, consist of a shaft which is keyed to a spring loaded drum. A slide contact is attached to one end of the shaft and this unit is able to rotate under spring tension within a case containing the resistance winding of a single turn 5000 ohm potentiometer. To the other end of the shaft is connected a circular disc to which a tension cable is attached. The tension cable is then attached to the pushrod or control cable whose displacement is to be measured. The range of displacement measured by the potentiometer is governed by the circumference of the circular disc. Two of these transducers were assembled and installed. One of the transducers determines the angular position of the aileron by connecting it to the aileron control rod which extends across the aircraft fuselage. This installation is shown in Figure 30. The other transducer was connected to the right rudder control cable, just forward of the rudder autopilot servo, to determine the angular position of the rudder. This installation is shown in Figure 31.

A five volt dc power supply, activated by the aircraft's 115 volt, 400 Hertz main inverter, provides a power source for the three control surface position transducers. The output of the five volt dc power supply is modified by a ground and two adjustable potentiometers so that the three transducers connected in parallel are provided with a voltage range of $\pm$ 1.4 volts dc. A schematic of the electrical circuit used to determine control surface position is shown in Figure 32.

Calibration curves relating the angular positions of the elevators, the ailerons, and the rudder to signal outputs were obtained by displacing the trailing edge of the appropriate control surface and recording the resulting signal. The elevators were locked in the neutral position by means of the gust lock system and the sliding arm of the potentiometer was positioned on the elevator pushrod so that the signal output was zero. The gust lock system was then released and the trailing edges of the elevators were displaced in increments of five degrees from fifteen degrees down to twenty-five degrees up. Trailing edge displacement was measured by a protractor situated so that its center coincided with the elevator hinge line. The resulting signal outputs were obtained from an electronic digital voltmeter that was connected to the signal terminal board. The same procedure was followed for the ailerons which have a trailing edge displacement that varies from fifteen degrees down to twenty degrees up, and for the rudder which has a trailing edge displacement that varies from twenty-one degrees left to twenty-one degrees right.

The calibration curve for the elevators, shown in Figure 33, indicates that the relation between the elevator position and the signal output is linear. It also shows that the slope is steeper when the trailing edges of the elevators are displaced downward than when the trailing edges are displaced upward. This is because the potentiometer is connected to a pushrod forward of the elevator bellcrank assembly.

The calibration curve for the ailerons is shown in Figure 34. It indicates a slightly nonlinear relationship between aileron position and signal output. This is due to the potentiometer being mounted on the aileron control rod which is forward of a number of pushrod and bellcrank assemblies.

The calibration curve for the rudder is shown in Figure 35. It indicates a linear relationship between rudder position and signal output, however the slope is slightly steeper when the trailing edge of the rudder is displaced to the left. The potentiometer is placed in this case so that the slope should be constant throughout the full range of rudder travel. When the trailing edge of the rudder is displaced to the left the circular disc attached to the control cable is positioned by spring tension. To ensure that the difference in slope was not a result of inadequate spring tension in the transducer, this spring tension was increased. The slope of the curves remained as shown in Figure 35. It is therefore assumed that there may be some slight misrigging or slack in the rudder control system.

3. Normal Acceleration

A Statham Model A5TC-8.0-350 accelerometer, shown in Figure 36, was utilized to measure the aircraft's normal acceleration. This accelerometer is designed to measure acceleration within a range from a negative two g's to a positive eight g's. It is of the unbonded strain gage bridge type which generally has the strain gages attached to a fixed frame and a force-summing member. When the force-summing member is displaced, the balance of the bridge is changed, thus providing an electrical output proportional to the magnitude of the applied force.

A strain gage signal amplifier of the type utilized in the control force electrical circuit was used with the accelerometer to provide a signal output varying from a negative 0.600 volts to a positive 1.400 volts dc. The polarity of the output was arranged so that a negative signal corresponded with a negative acceleration while a positive signal corresponded with a positive acceleration. A schematic of the electrical circuit used to measure normal acceleration is shown in Figure 37.

Static calibrations of the accelerometer were conducted by both the two-g-turnover method and by use of a centrifuge. To calibrate the accelerometer by means of the two-g-turnover method the transducer was placed on a level platform with its sensitive axis perpendicular to the earth's gravitational field. The balance of the strain gage amplifier was adjusted so as to provide a zero signal output for this condition of zero g's. The accelerometer was then rotated to the positive one-g position and the gain adjusted to provide a positive signal output of approximately 0.180 volts. This process was repeated until the desired balance and gain remained stable. This resulted in signal outputs of a negative 0.187, zero, and a positive 0.180 volts for a negative one, zero, and a positive one g's respectively.

To obtain a static calibration of the accelerometer at higher g-levels, the centrifuge shown in Figure 38 was utilized. This engine-driven centrifuge is equipped with instrumentation so that one-half its angular speed in revolutions per minute could be read on an electronic counter. In addition it is provided with slip rings so that the electrical signals generated by the accelerometer could be transferred via the signal amplifier to an electronic digital voltmeter. The accelerometer was attached to the centrifuge by means of a plate upon which the centrifuge's center of rotation and the distance from that center in half inch increments were scribed. This mounting, shown in Figure 39, was such that the accelerometer could be moved radially to varying distances from the centrifuge's center of rotation. To determine the accelerometer's center of mass, several test runs were made at varying speeds and the accelerometer was repositioned until its center of mass coincided with the centrifuge's center of rotation as indicated by a signal output of zero volts which remained constant with varying centrifuge speed. The accelerometer was then positioned so that its sensitive axis was aligned with a radius of rotation and its center of mass was one half inch from the centrifuge's center of rotation. The centrifuge was run at varying speeds which were recorded along with the accelerometer's corresponding signal output. The static acceleration generated by the centrifuge was calculated from the expression

$$a = \frac{4\pi^2 N^2 r}{g(12)\,(3600)} = 2.840 \times 10^{-5} N^2 r ,$$

where r equals the radius of rotation of the accelerometer's center of mass in inches, N equals the centrifuge speed in revolutions per minute, and a equals the acceleration in units of g's. The values thus obtained were plotted versus the corresponding signal output resulting in the calibration curve illustrated in Figure 40. This curve indicates a reasonably linear relationship until the design limits of a negative two and a positive eight g's are approached. At approximately seventy-five percent of the design limits, the slope of the calibration curve decreases indicating a possible shift in the accelerometer's center of mass. This could be due to the displacement of the accelerometer's force-summing member.

Upon completion of the accelerometer's static calibration the aircraft's center of gravity was computed in accordance with the procedures outlined in "AN 01-1B-40, Weight and Balance Data for Model US-2A Airplane, Serial No. 136533." The center of gravity was found to vary only slightly from a most forward position located at Fuselage Flight Station 211 to a most aft position located at Station 214. The aircraft's fuselage flight station locations are shown in Figure 41. The accelerometer was installed along the aircraft's center line at Station 212.5. This installation is shown in Figure 42.

V. PREFLIGHT PROCEDURES

The preflight procedures which should be performed prior to each test flight consist of: (1) a systematic checkout of the magnetic tape recorder and (2) checking the sensing transducers for the proper balance and gain.

An external dc power source may be connected to the aircraft's electrical system to provide the necessary electrical power while the preflight procedures are being conducted. This external source activates the aircraft's dc bus system and main inverter thus providing the required 28 volts dc and 115 volts, 400 Hertz, for the data acquisition system.

A checkout of the magnetic tape recorder may be conducted by utilizing the test unit shown in Figure 11. The only check required on the test unit is meter zeroing. If the meter needle does not read zero, the zeroing screw on the meter panel may be adjusted to obtain the proper reading. The testing unit is then connected to the test unit receptacle on the amplifier cabinet and the power switches on the remote control unit and the test unit are turned on. The equipment should be allowed to warm up for at least one-half hour. Upon completion of the warm-up period the meter switch is rotated through the following positions: 28 V DC, 75 V DC, 150 V DC, 250 V DC, 115 V AC, and 12.6 V AC. The meter needle should fall between the two limits on the meter scale thus indicating that the system voltages are within the proper range. The carrier head current of the FM amplifiers is checked by placing the METER switch in the FM position and the TRACK switch in the numbered position which corresponds to the track occupied by the FM amplifier to be checked. The RECORD button is then pushed. The needle should fall at center scale or above.

The bias level of the oscillator in the electronics power supply is checked by placing the METER switch in the BIAS position and the TRACK switch in the numbered position which corresponds to the track occupied by the AM amplifier. With the signal input cable disconnected, the meter needle should read within the limits on the meter scale. The signal head current of the AM amplifier is checked by placing the METER switch in the SIGNAL position and the TRACK switch in the numbered position which corresponds to the track occupied by the AM amplifier. With the signal input cable to the AM amplifier disconnected, adjust the BIAS FILTER BALANCE, C and R controls on the test unit, for a minimum meter reading. A 1000 Hertz signal is then connected to the AM amplifier and the meter needle observed. It should fall between the two limits on the meter scale.

If the meter needle falls outside the required limits, it indicates that the component checked is either malfunctioning or out of adjustment. To get proper performance out of the data acquisition system these discrepancies will have to be corrected. The bias level of the oscillator in the electronics power supply may be adjusted by means of the HEAD BIAS ADJUST screw located on the electronics power supply. The signal head current of the AM amplifier may be adjusted by means of the RECORD LEVEL control on the amplifier. If discrepancies are found in the system voltages or the carrier head current of the FM amplifiers, repairs or adjustments will have to be made. These fall outside the scope of the preflight procedures.

The balance and gain of the sensing transducers may be checked with the aid of an electronic digital voltmeter. With no loads applied to the control wheel force transmitter or the rudder pedals, the digital voltmeter, when connected to the corresponding terminal on the signal terminal board, should read zero volts. This indicates a proper balance. The gain may be checked by applying a measured force to the control wheel or rudder pedals by means of a calibrated dial push-pull gauge and then comparing the voltmeter signal output with those shown on the corresponding calibration curves. The two-way polarity switch utilized in the aileron and the rudder control force circuits should be in the up position. If the proper signals are not obtained, the balance and gain may be adjusted by means of the BAL and GAIN control on the corresponding strain gage signal amplifier.

By engaging the aircraft's gust lock system the control surfaces may be locked in the neutral position. The digital voltmeter, when connected to the proper terminal on the signal terminal board, should then indicate a reading of zero volts if the sliding arm or circular disc of the potentiometer is properly positioned. The aircraft's gust lock system may then be released and the control surfaces deflected to their stops. The digital voltmeter should provide an output signal that corresponds with the full deflection signal shown on the calibration curve. If the proper signals are not obtained the balance may be adjusted by positioning the sliding arm or the circular disc of the potentiometer so that it is connected to its pushrod or control cable in such a manner that a zero signal output is obtained when the control surface is in the neutral position. The gain may be corrected by adjusting the two 1000-ohm potentiometers shown in Figure 32 to ensure that they provide the proper voltage range of $\pm$ 1.4 volts dc across the three potentiometers connected in parallel.

The balance and gain of the accelerometers may be checked by performing a static calibration by means of the two-g-turnover method. With a digital voltmeter connected to the proper terminal on the signal terminal board, signal outputs of a negative 187, zero, and a positive 180 volts should be obtained for the negative one-, zero-, and positive one-g positions, respectively. If the proper signals are not obtained the balance and gain may be adjusted by means of the BAL and GAIN control on the corresponding strain gage signal amplifier.

When the system checkout is completed the tape reel may be mounted in the tape transport and the co-axial cables from the signal terminal board may be attached to the desired amplifier connector at the rear of the record amplifier cabinet. Data sensed by the transducers may now be recorded by activating the POWER and RECORD switches located on the remote control unit.

VI. SYSTEM EVALUATION

The requirements specified for the data acquisition system installed in the US-2A aircraft necessitated a system that (1) could be procured at little cost, (2) could be installed and calibrated using available personnel and facilities, (3) could be operated without unduly complex preflight procedures, and (4) would provide a reasonable degree of reliability and accuracy.

The magnetic tape recorder was obtained on a loan basis. The data sensing transducers and the associated electronic equipment necessary to complete the required electrical circuits were acquired as surplus equipment from other organizations or from stocks at the Naval Postgraduate School. Thus the cost of the installation has been minimal.

Except for the installation of the strain gage bridge attached to the co-pilot's rudder pushrods and some electrical wiring, the installation and calibration of the data acquisition system was conducted by personnel attached to the Aeronautics Department of the Naval Postgraduate School using facilities available to the School.

Preflight procedures are not unduly complicated. However they are time consuming particularly because of the warmup period. This lengthy warm-up period plus the time required to complete the system checkout necessitates the operation of an engine-driven external power source for an extended period of time. As these external power sources are utilized for the launching of aircraft assigned to the Naval Auxiliary Landing Field, Monterey, California, the preflight procedures will have to be conducted during a period when use of this equipment will not interfere with flight operations.

The reliability and accuracy of the system must be evaluated in terms of the tolerances allowed for a given project. The accuracy of the system is considered to be well within the limits required of a flying classroom. Calibration curves are provided for the control forces, the control surface positions and the normal acceleration. While these curves are not linear in all cases they may be interpolated so that the signal output can be related to a given control force, control surface position, or normal acceleration. The degree of accuracy obtained from these curves will be dependent on the scale used to graph them. The data in Table I may be plotted to provide a calibration curve between the signal input to a particular FM amplifier and the signal output obtained from the Ampex Series FR1100 Magnetic Tape Record/Reproduce unit. By utilizing both the calibration curve for the FM amplifier and the calibration curve for the corresponding sensing transducer a reasonable degree of accuracy should be obtained.

The system does have minor defects which tend to limit the degree of accuracy and flexibility that can be obtained. The control wheel force transmitter employs only two cantilever beams. These beams which are deflected fore or aft to measure elevator control forces and up or down to measure aileron control forces can also be subjected to a twisting motion. The strain gage installation on the cantilever beams eliminates any cross-coupling between the fore or aft and up or down bending. However any twisting of the cantilever beams will result in an erroneous signal output. Tests conducted during the calibration of the control wheel force transmitter indicated that a twisting moment of five foot-pounds resulted in an error of approximately 0.30 volts, irrespective of the bending forces applied. This error may be reduced if the control wheel forces are applied equally at the center of the two control wheel handles without a twisting motion.

A single transducer was utilized to measure the position of both ailerons by connecting it to the aileron control rod which extends across the fusalage. While this installation allows for measurement of the angular position of both ailerons by placing the transducer equally distant from each aileron, it also places the transducer forward of a number of pushrod and bellcrank assemblies. This may result in a time-lag effect which will decrease the accuracy of the aileron position measurements.

Two-way strain gage signal amplifiers were not available. This limits the flexibility of the system as it necessitated the incorporation of a switching device in the aileron and rudder control force circuits if the maximum force levels in both directions are to be measured.

VII. CONCLUSIONS AND RECOMMENDATIONS

The instrumentation that was installed in the US-2A aircraft represents the initial phase in the efforts of the Aeronautics Department to procure an airborne data acquisition system. The installation is not perfect. It has some limitations, but these are minor. If the system is used properly it should provide excellent results with a high degree of accuracy. The system is flexible and has growth potential. The signal terminal board and corresponding co-axial cables have a thirteen signal capacity. The magnetic tape recorder can process six channels of high frequency aerodynamic data and one channel of wide-band direct voice recording for tape annotation. The co-axial cables attached to the signal terminal board may be connected or disconnected from the tape recorder in flight so that more than six aerodynamic parameters may be recorded during any one flight period without multiplexing.

For the system to achieve its full potential, additional equipment should be installed. It is recommended that the next phase in this project be the installation and calibration of data sensing devices useful in control and stability flight testing which are compatible with the present system. This installation should include devices that measure angle of attack, angle of sideslip, airspeed, altitude, pitch rate, roll rate, and yaw rate.

Future plans should include the installation of a photo panel and sensitive instrument gages that could be utilized in performance testing.

APPENDIX A

FIGURES

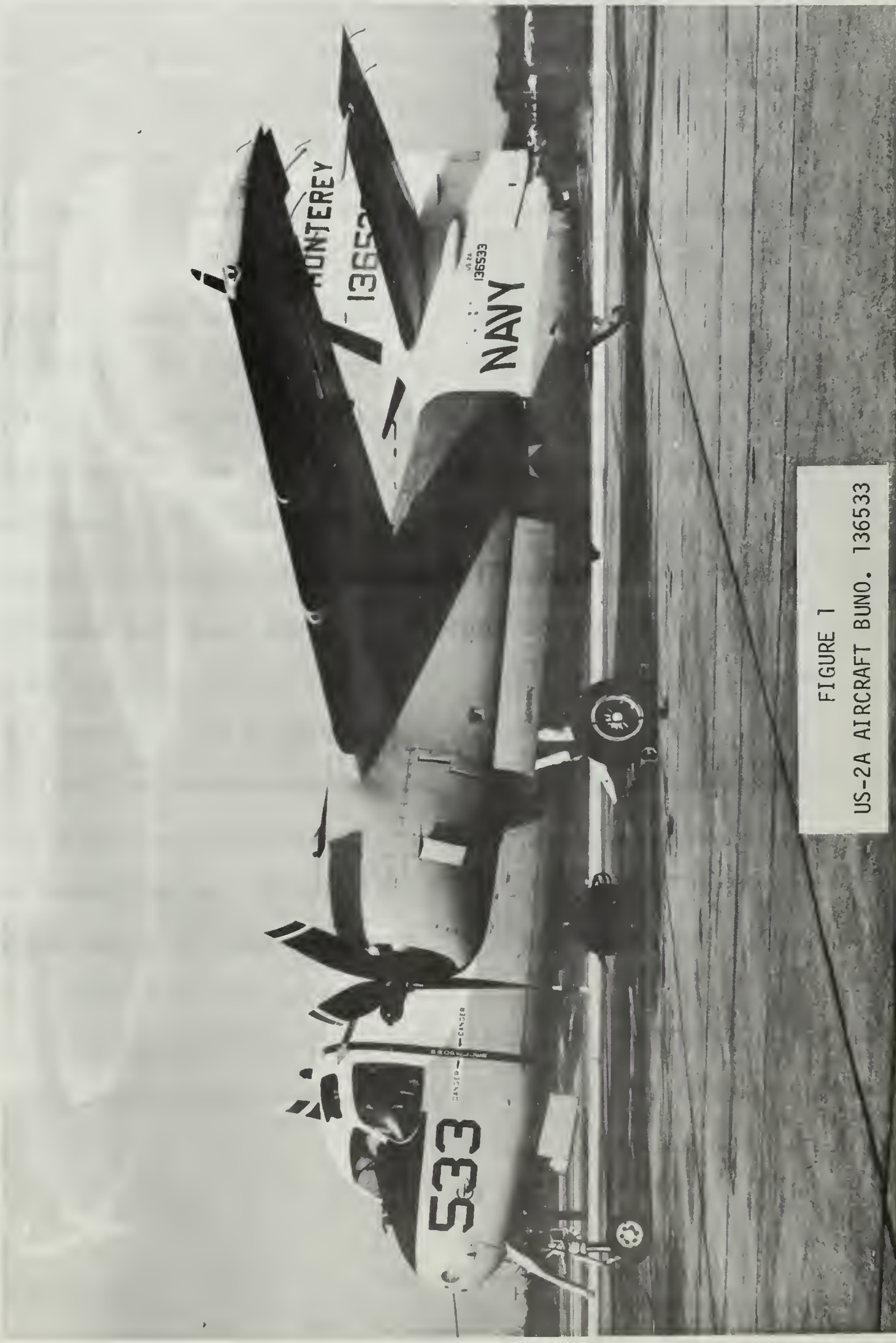


FIGURE 1
US-2A AIRCRAFT BUNO. 136533

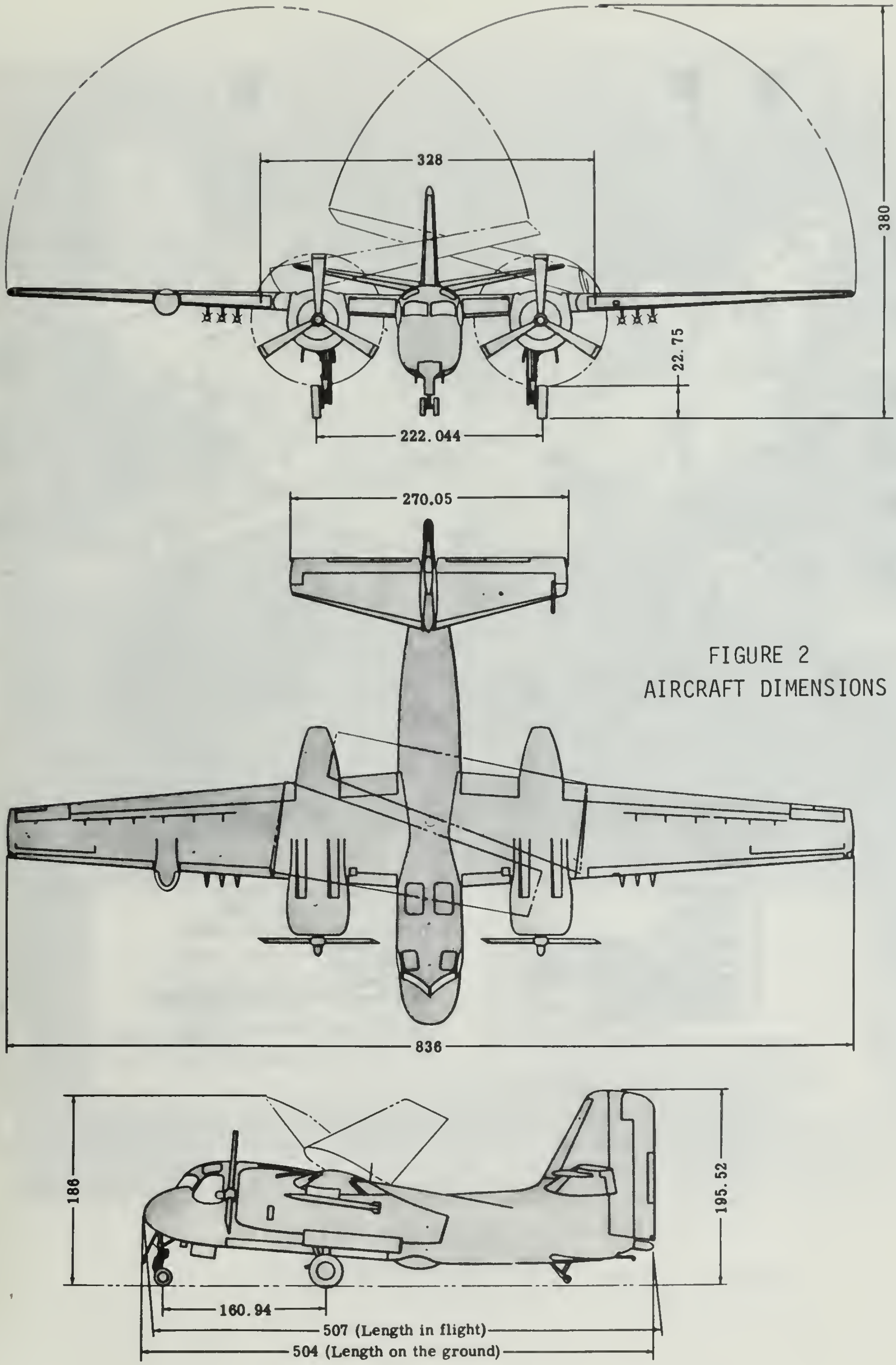
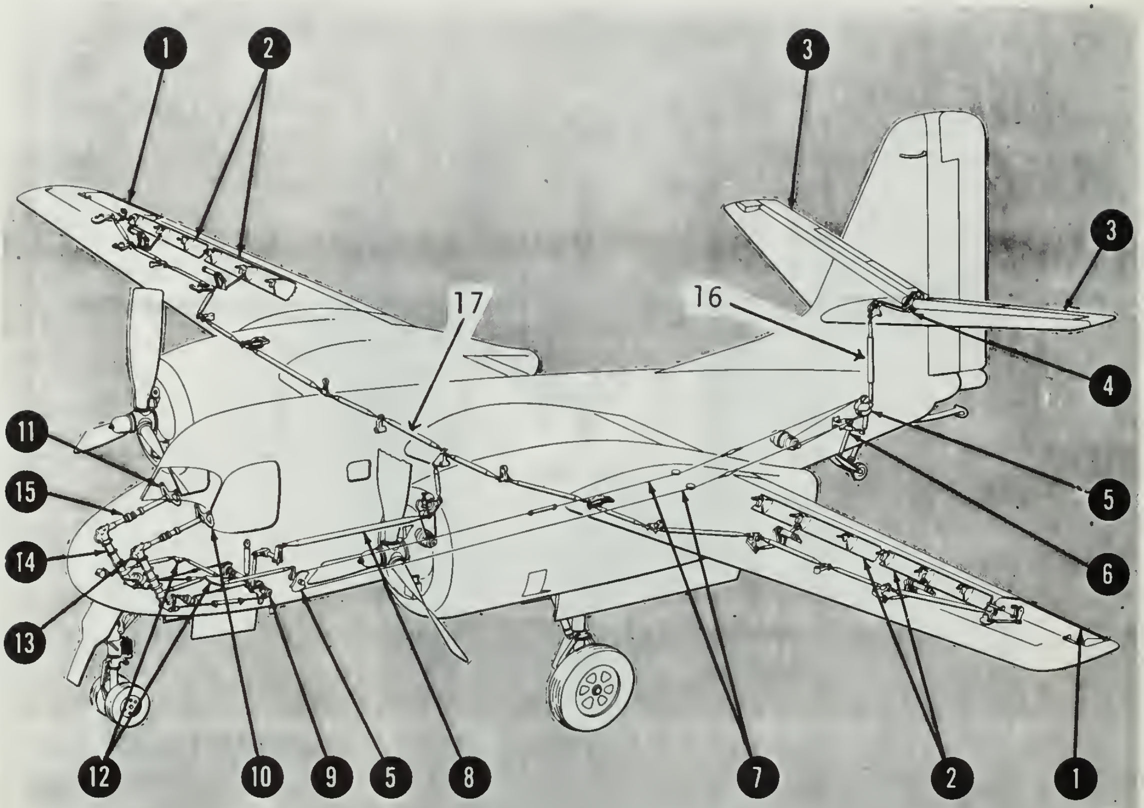


FIGURE 2
AIRCRAFT DIMENSIONS



1. AILERON
2. SPOILERS
3. ELEVATOR
4. BELLCRANK ASSEMBLY
5. ELEVATOR CONTROL SECTOR
6. DOWN-SPRING MECHANISM
7. CONTROL CABLE
8. PUSHROD ASSEMBLY
9. SECTOR TUBE ASSEMBLY
10. PILOT'S CONTROL WHEEL
11. CO-PILOT'S CONTROL WHEEL
12. BELLCRANK
13. SPROCKET
14. CONTROL COLUMN
15. CO-PILOT'S CONTROL DISCONNECT
16. ELEVATOR POSITION TRANSDUCER
17. AILERON POSITION TRANSDUCER

FIGURE 3

AIRCRAFT LONGITUDINAL AND LATERAL FLIGHT CONTROL SYSTEMS

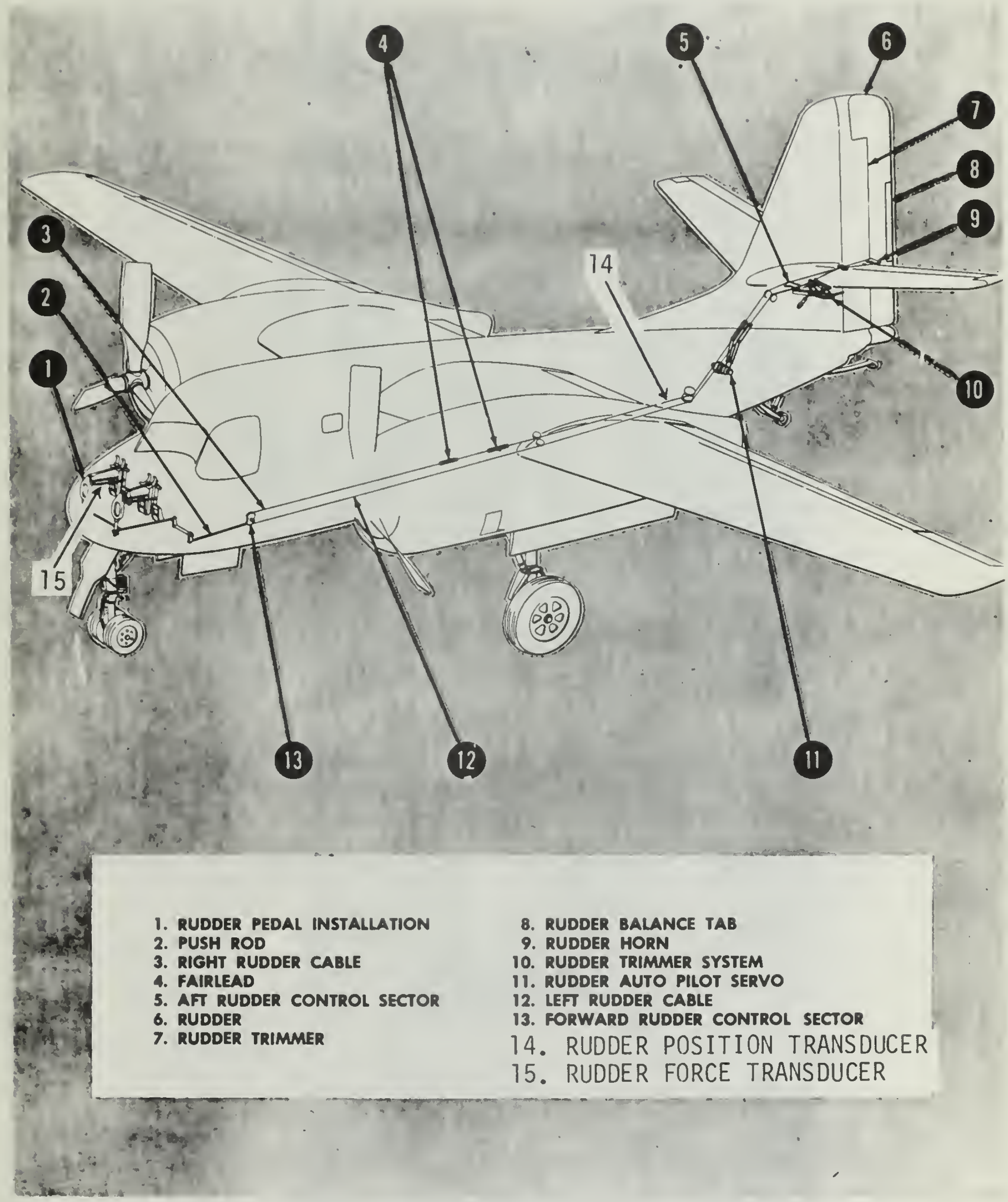


FIGURE 4

AIRCRAFT DIRECTIONAL FLIGHT CONTROL SYSTEM

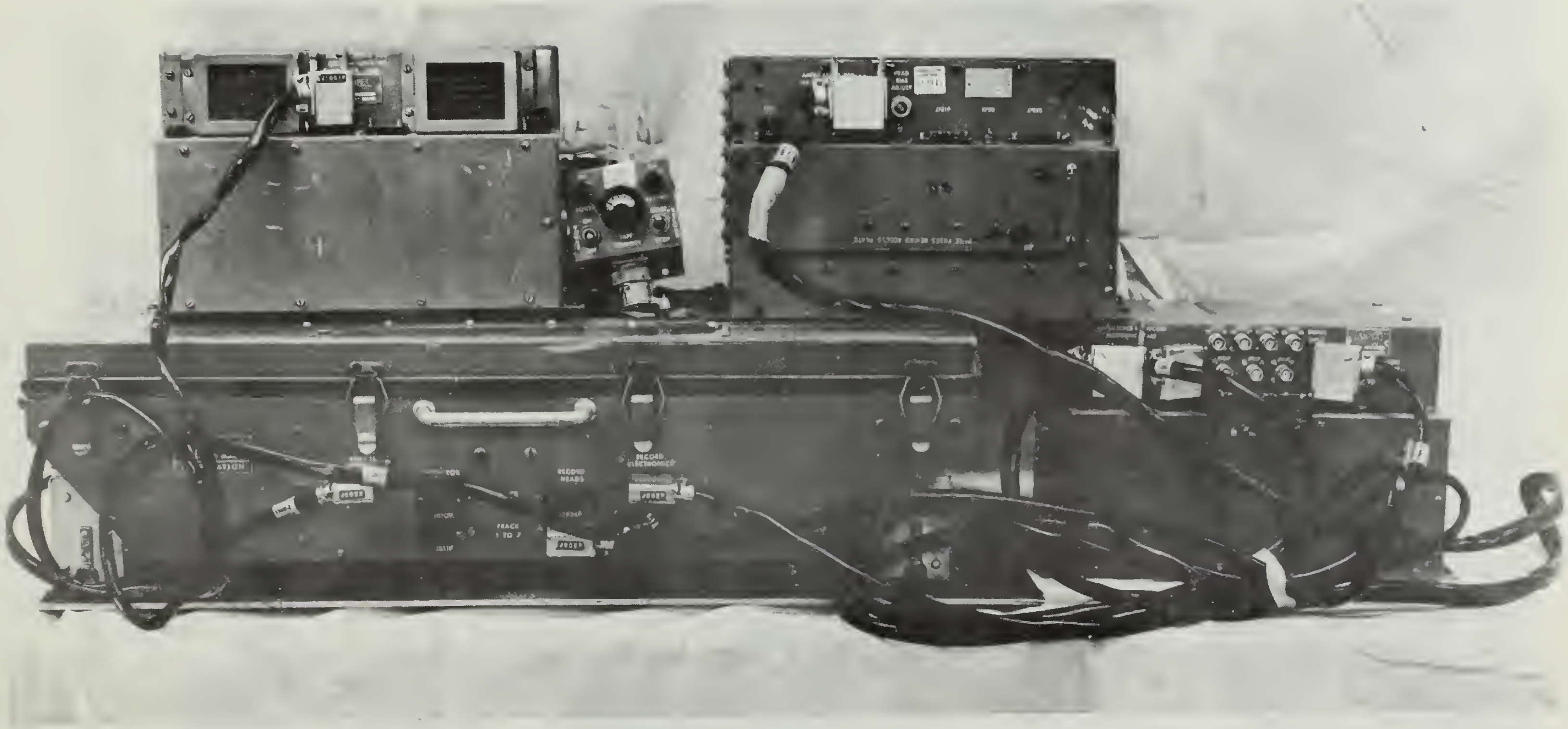


FIGURE 5
AMPEX SERIES 800 MAGNETIC TAPE RECORDER

FIGURE 6
TAPE TRANSPORT MECHANISM

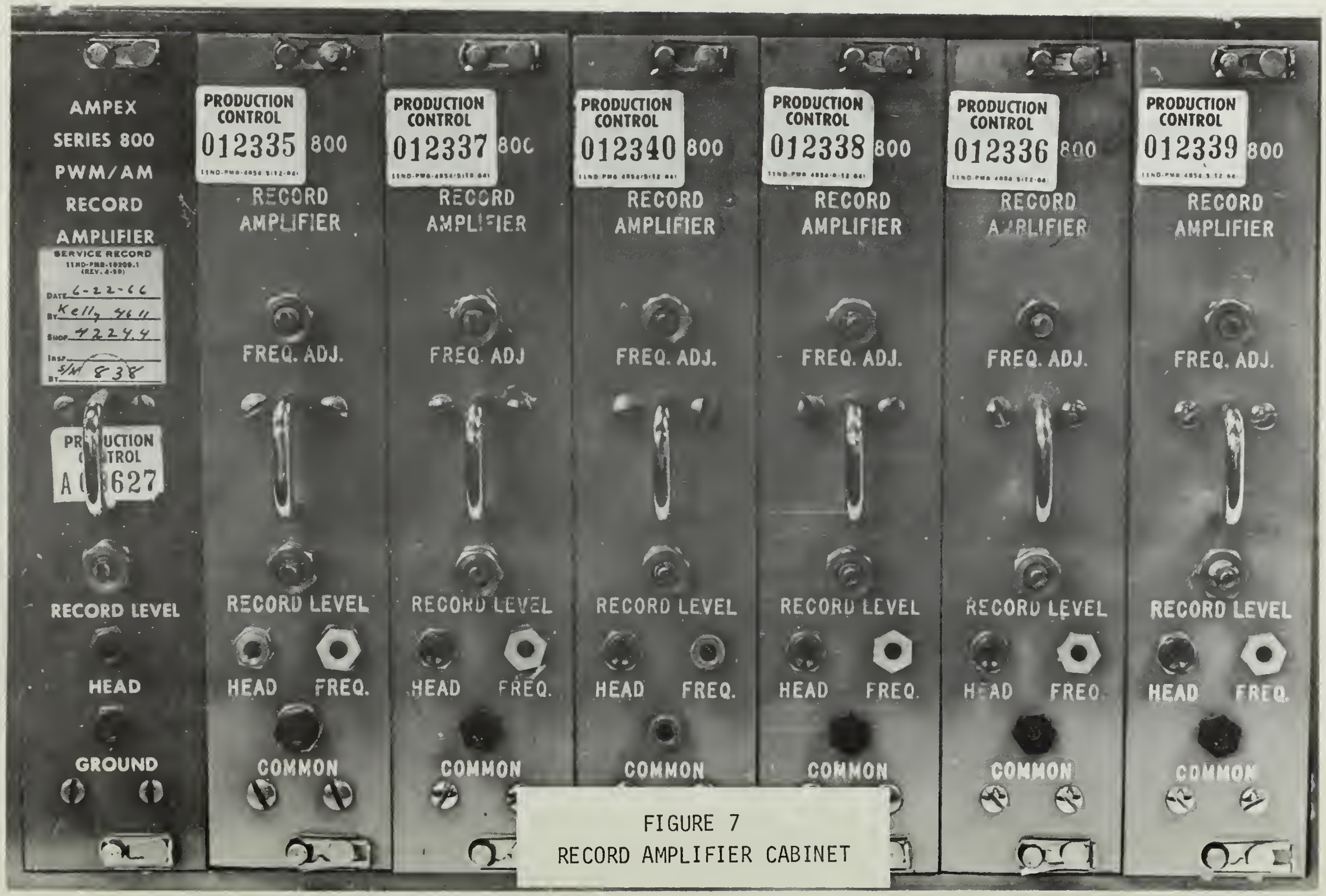


FIGURE 7
RECORD AMPLIFIER CABINET

FIGURE 8
ELECTRONICS POWER SUPPLY

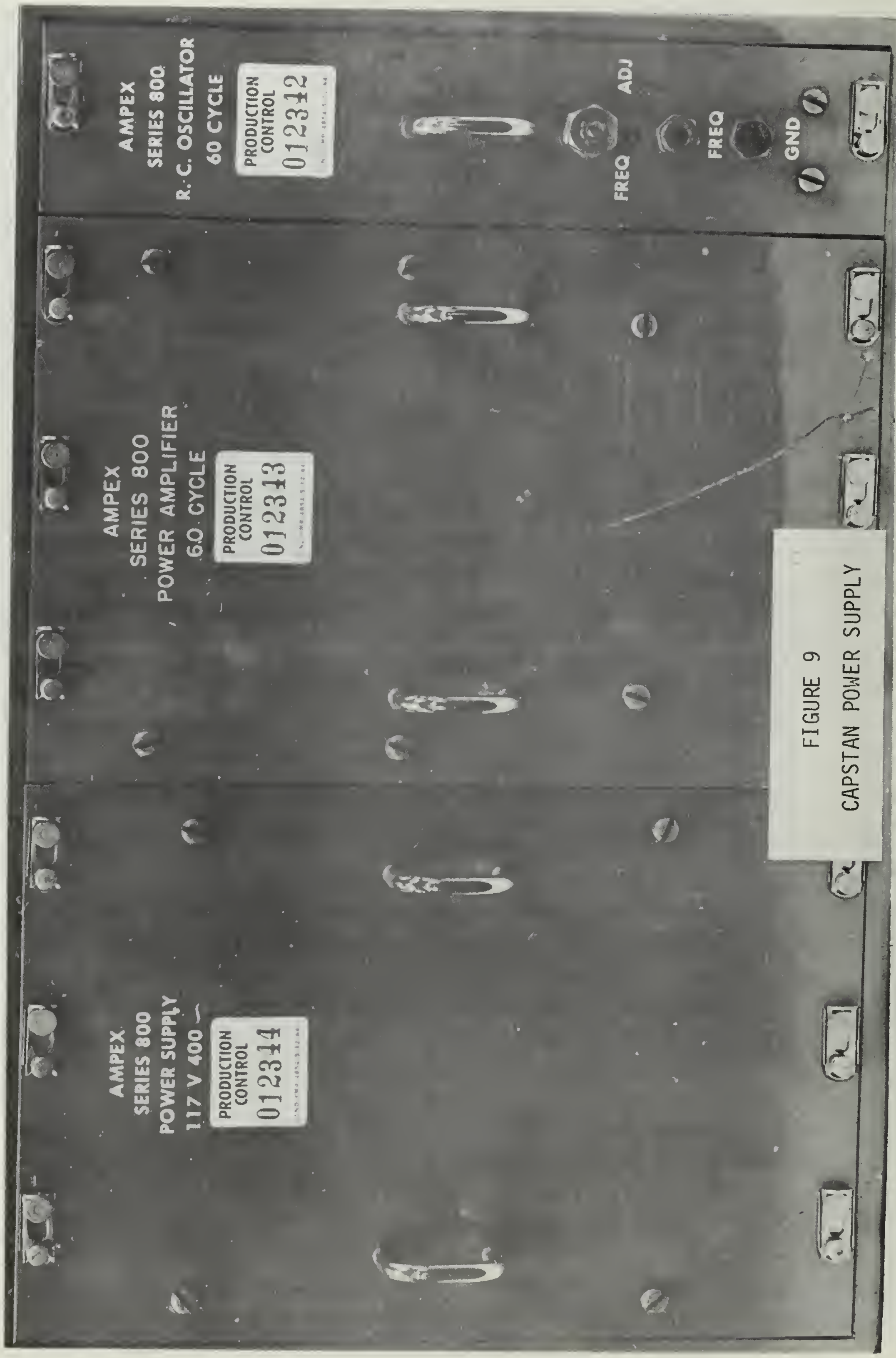


FIGURE 9
CAPSTAN POWER SUPPLY

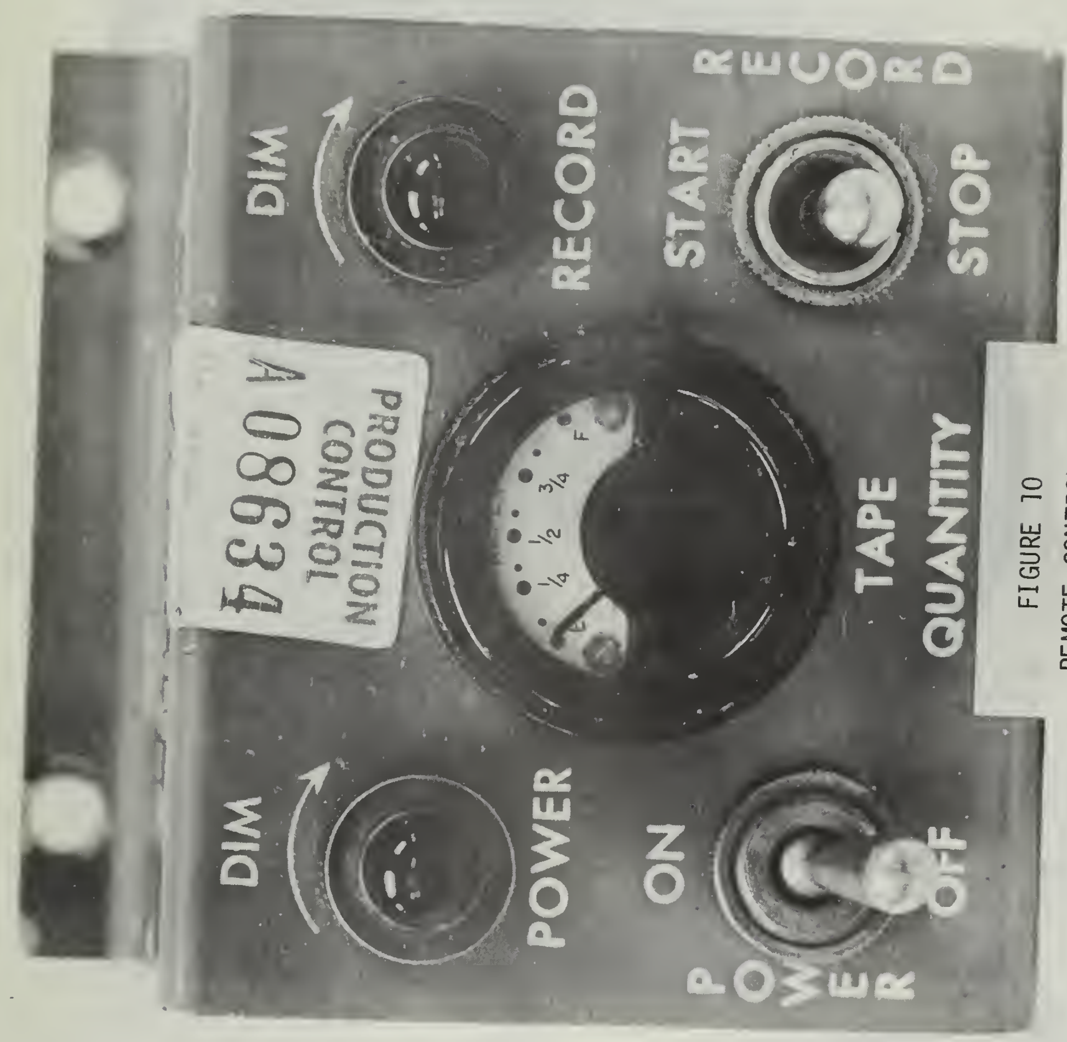


FIGURE 10
REMOTE CONTROL UNIT

FIGURE 11
RECORDER TEST UNIT

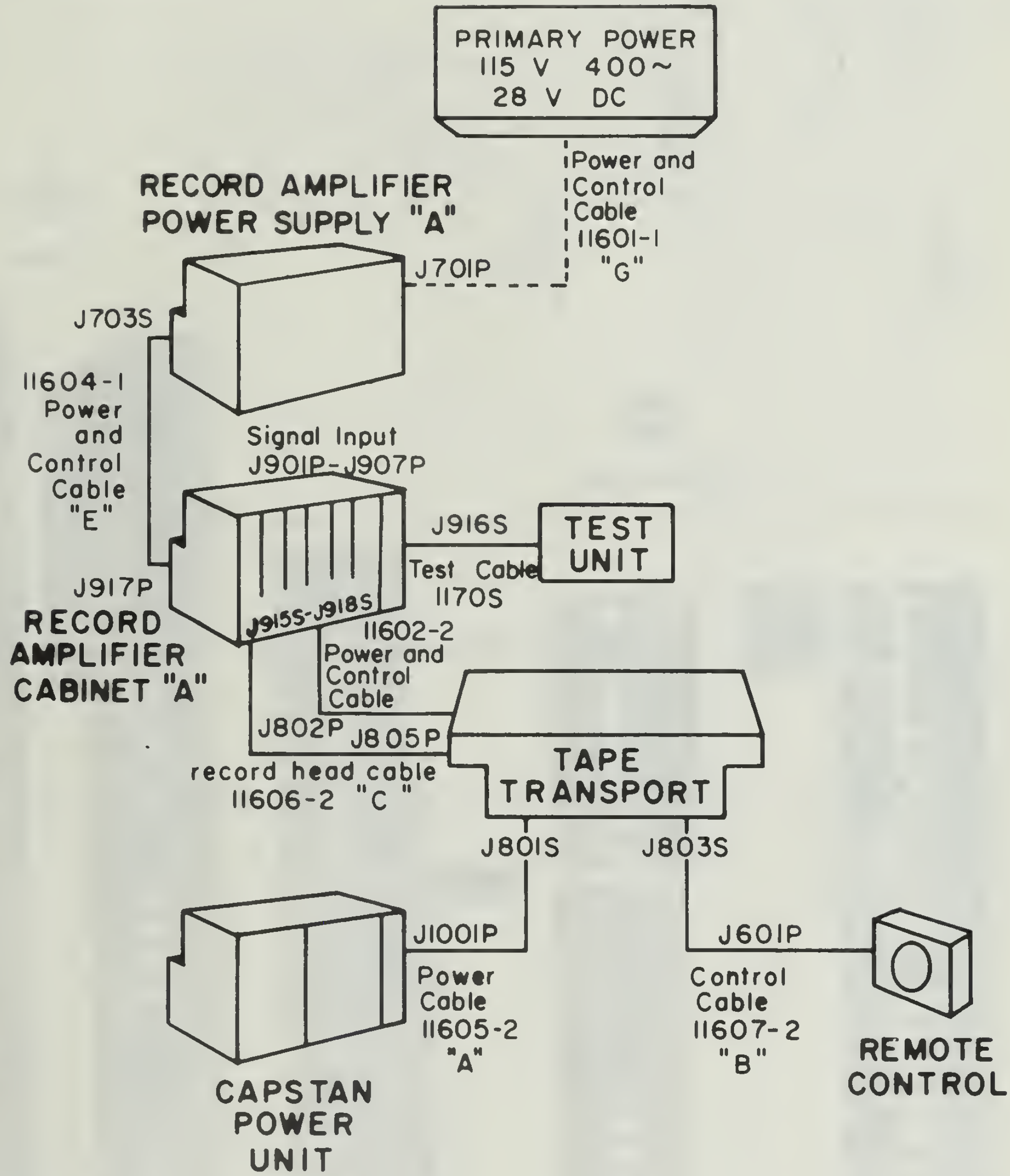


Figure 12
Cabling Diagram

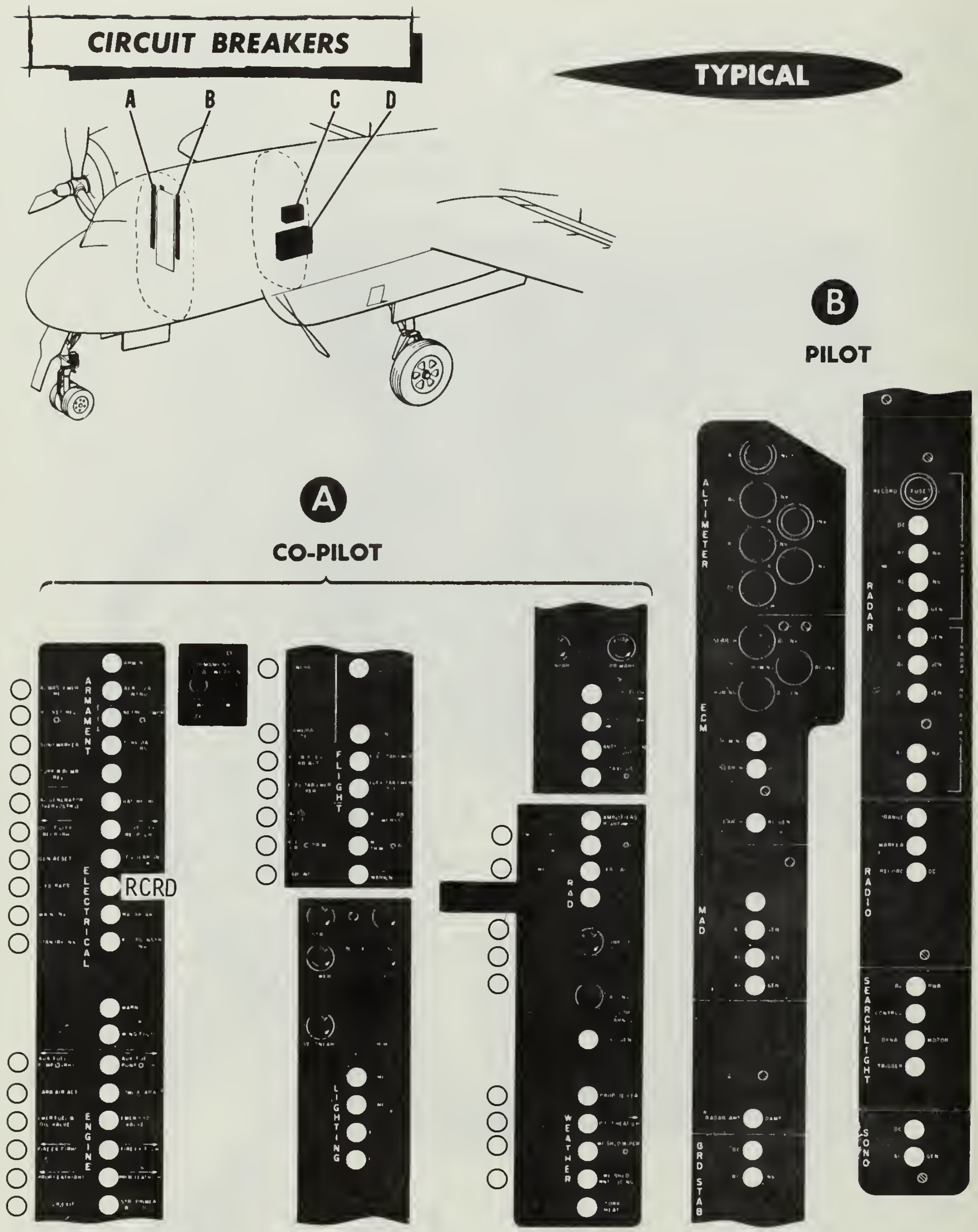
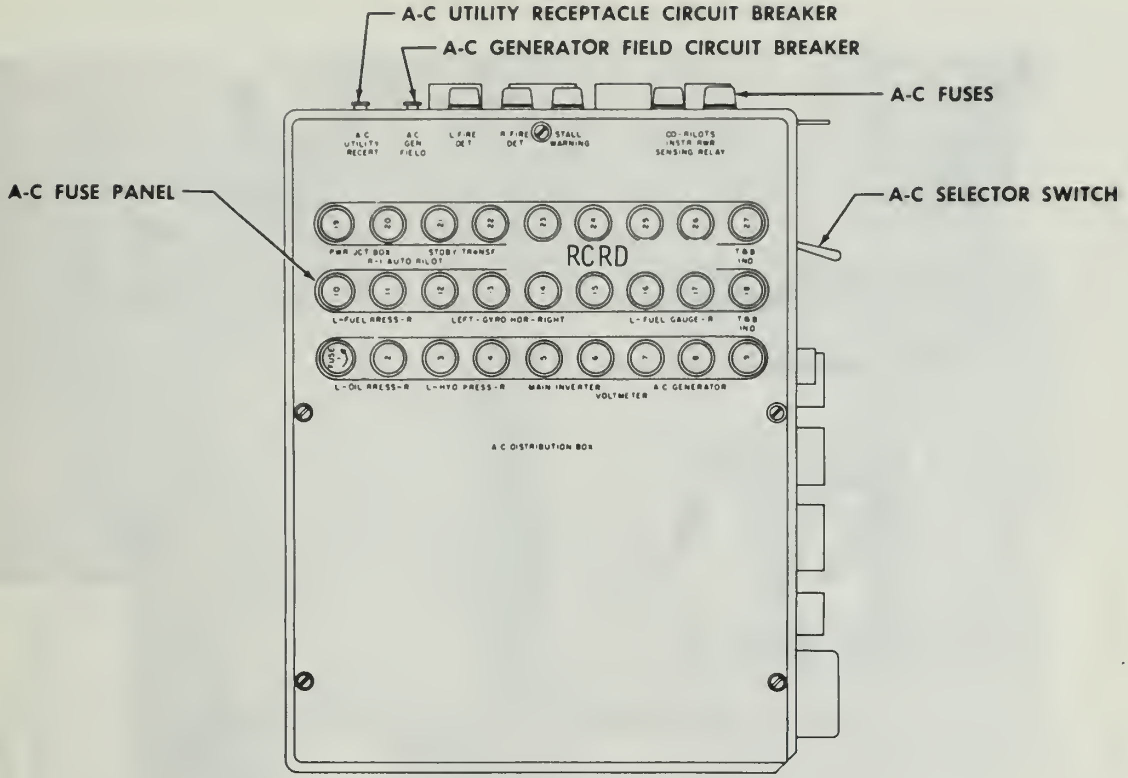


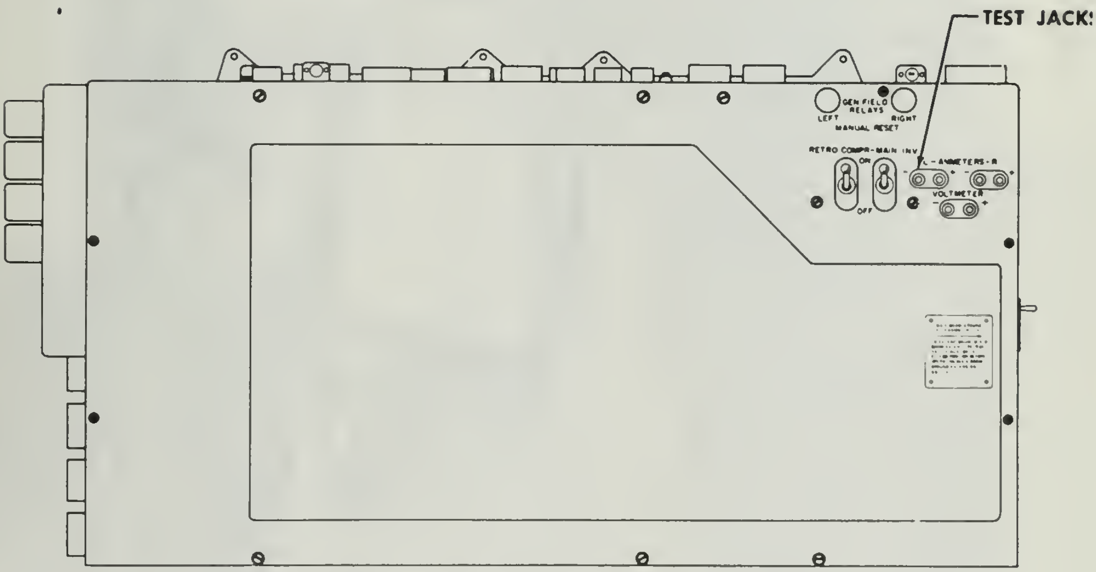
FIGURE 13

AIRCRAFT ELECTRICAL SYSTEM

CIRCUIT BREAKER PANELS



C **A-C DISTRIBUTION BOX**



D **D-C MAIN DISTRIBUTION BOX**

FIGURE 14

AIRCRAFT ELECTRICAL SYSTEM DISTRIBUTION BOXES

FIGURE 15
AMPEX SERIES FR1100 MAGNETIC TAPE
RECORD/REPRODUCE UNIT

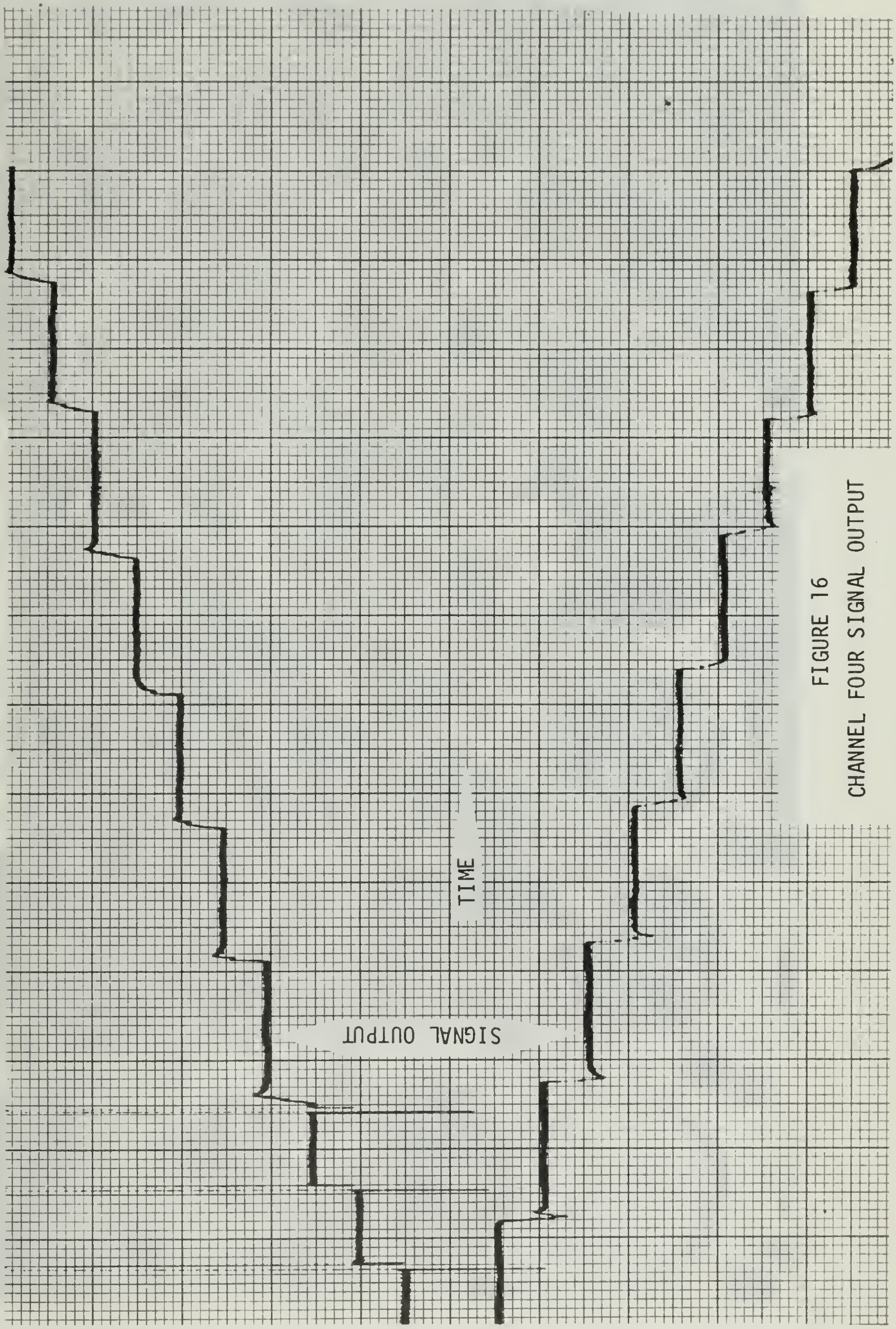


FIGURE 16
CHANNEL FOUR SIGNAL OUTPUT

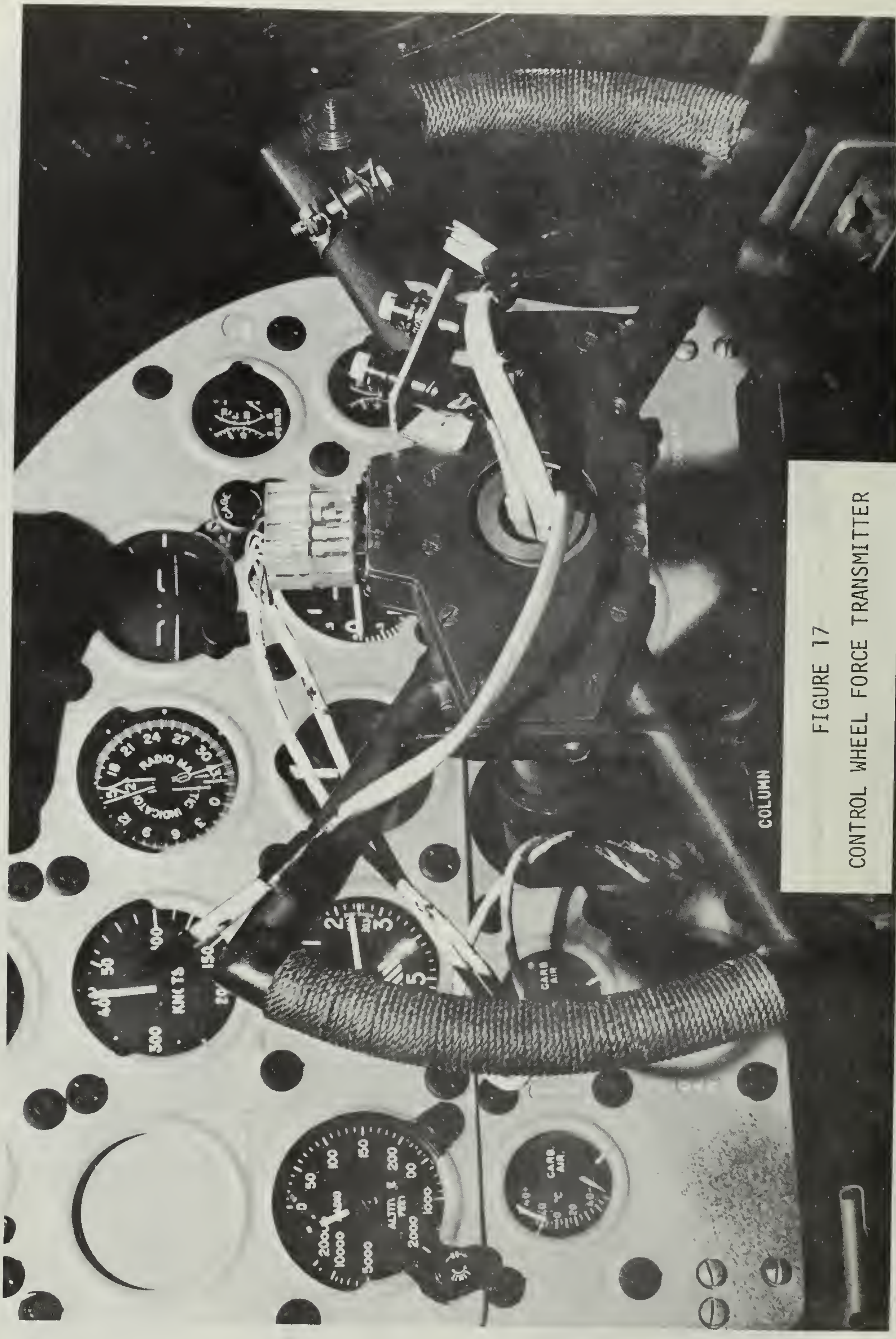


FIGURE 17
CONTROL WHEEL FORCE TRANSMITTER

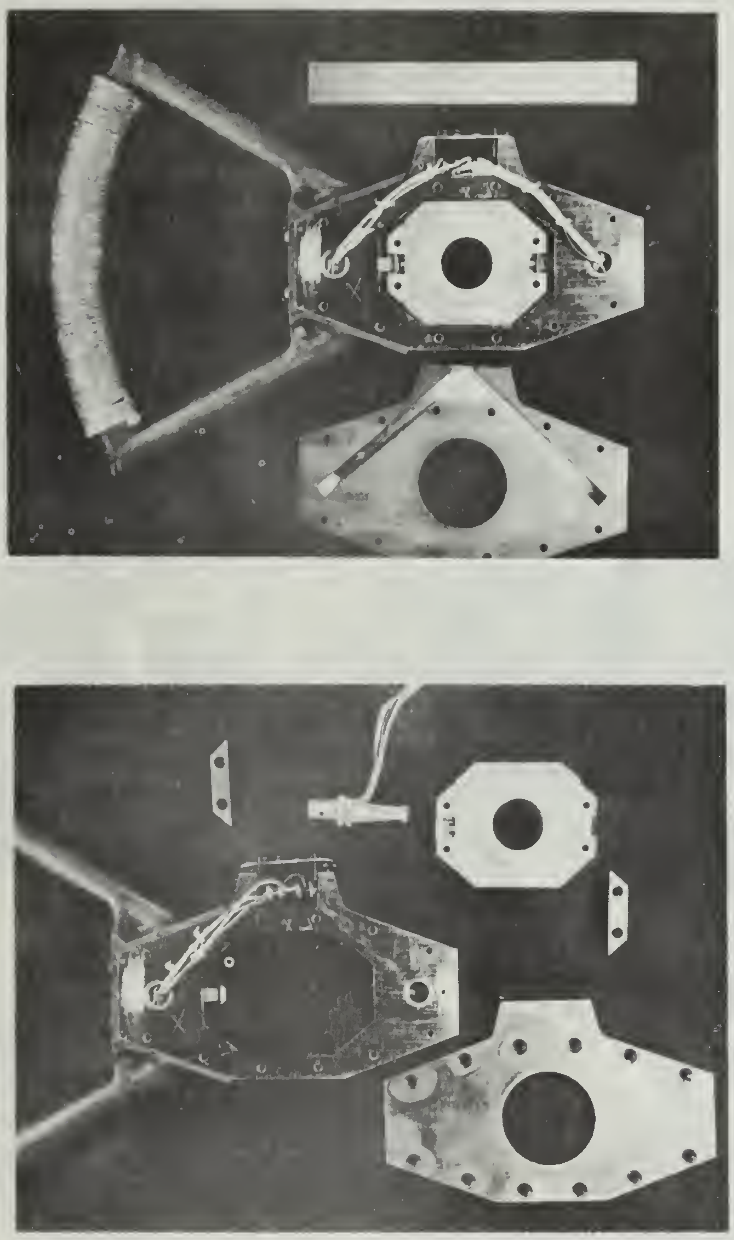


FIGURE 18
INTERNAL MECHANISMS OF THE CONTROL
WHEEL FORCE TRANSMITTER

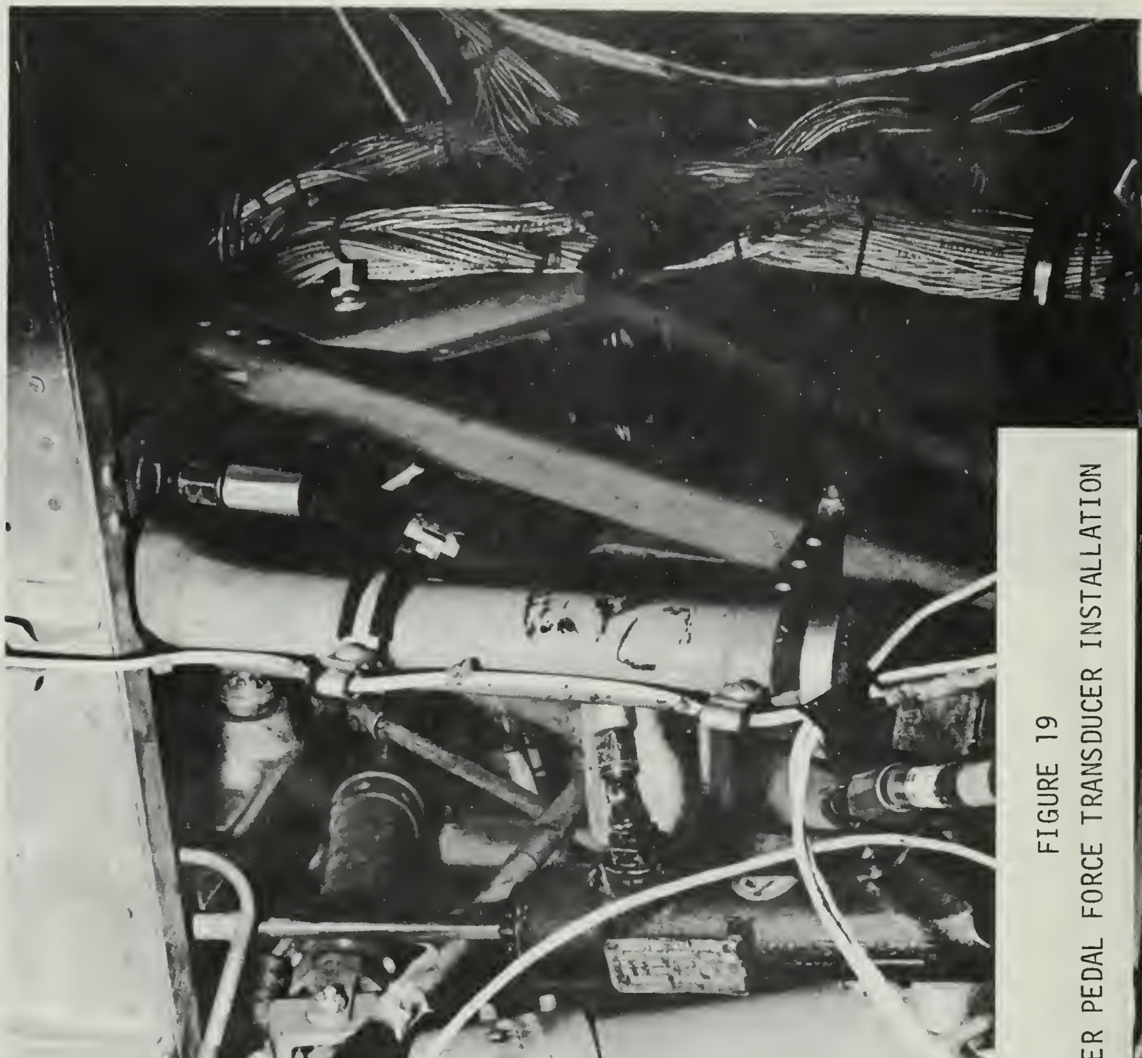


FIGURE 19
RUDDER PEDAL FORCE TRANSDUCER INSTALLATION

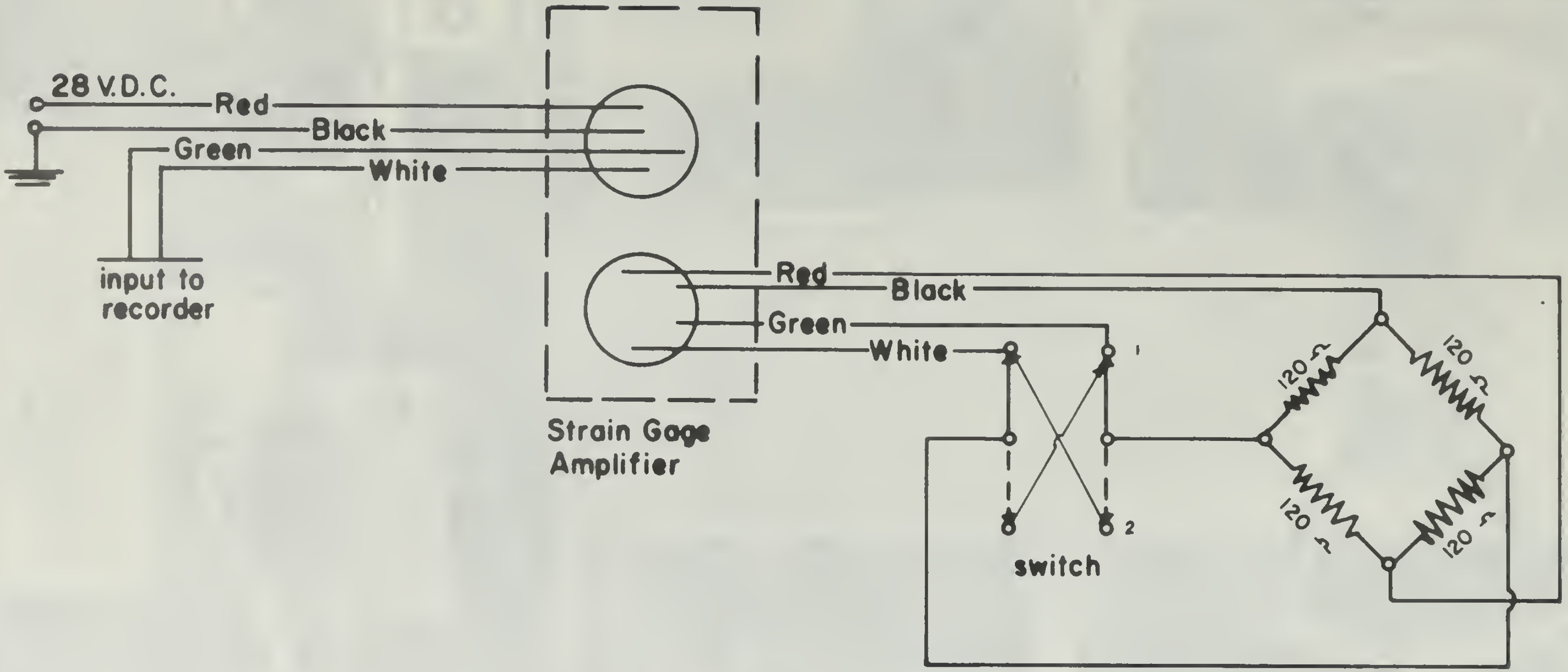


Figure 20

Control force schematic

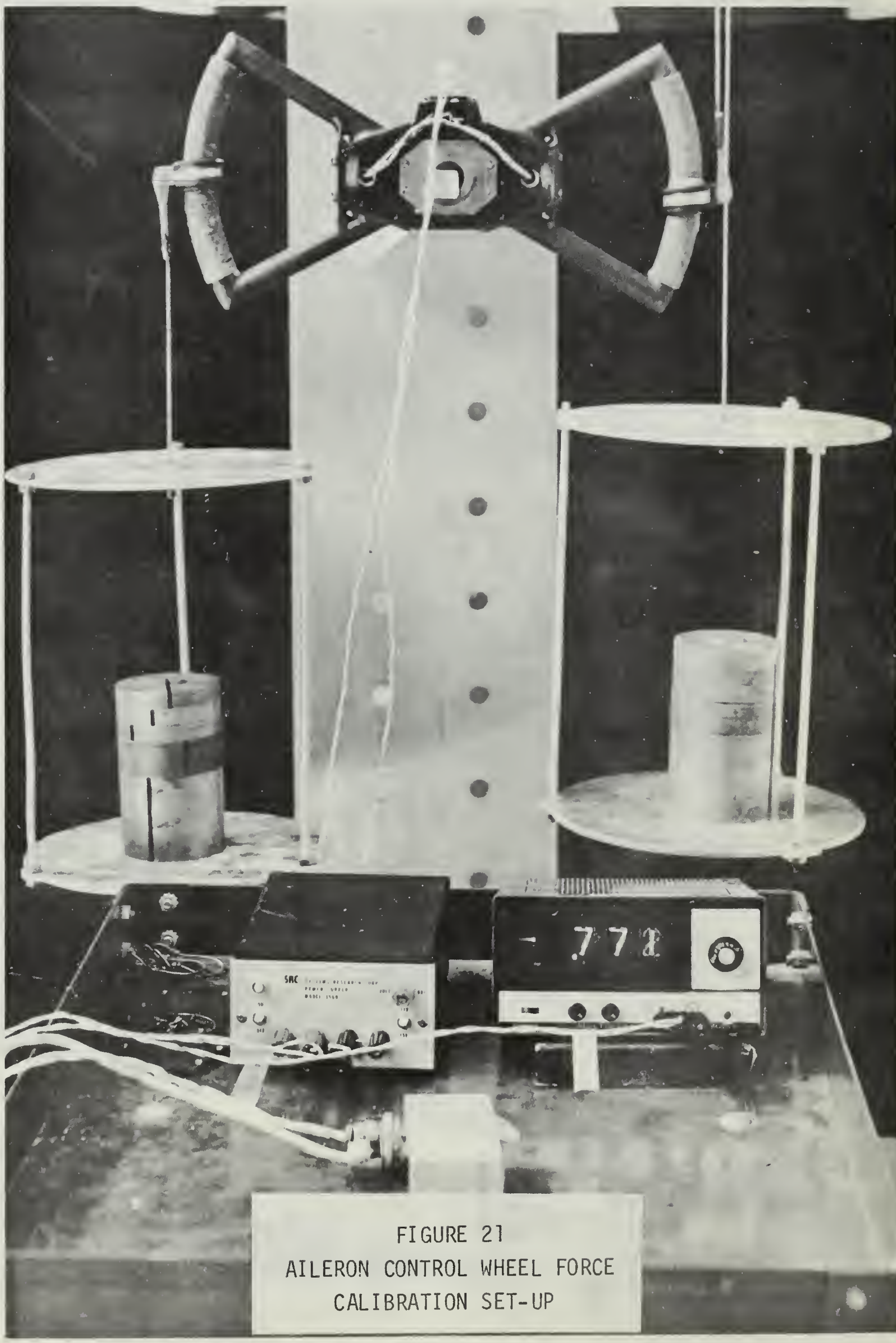


FIGURE 21
AILERON CONTROL WHEEL FORCE
CALIBRATION SET-UP

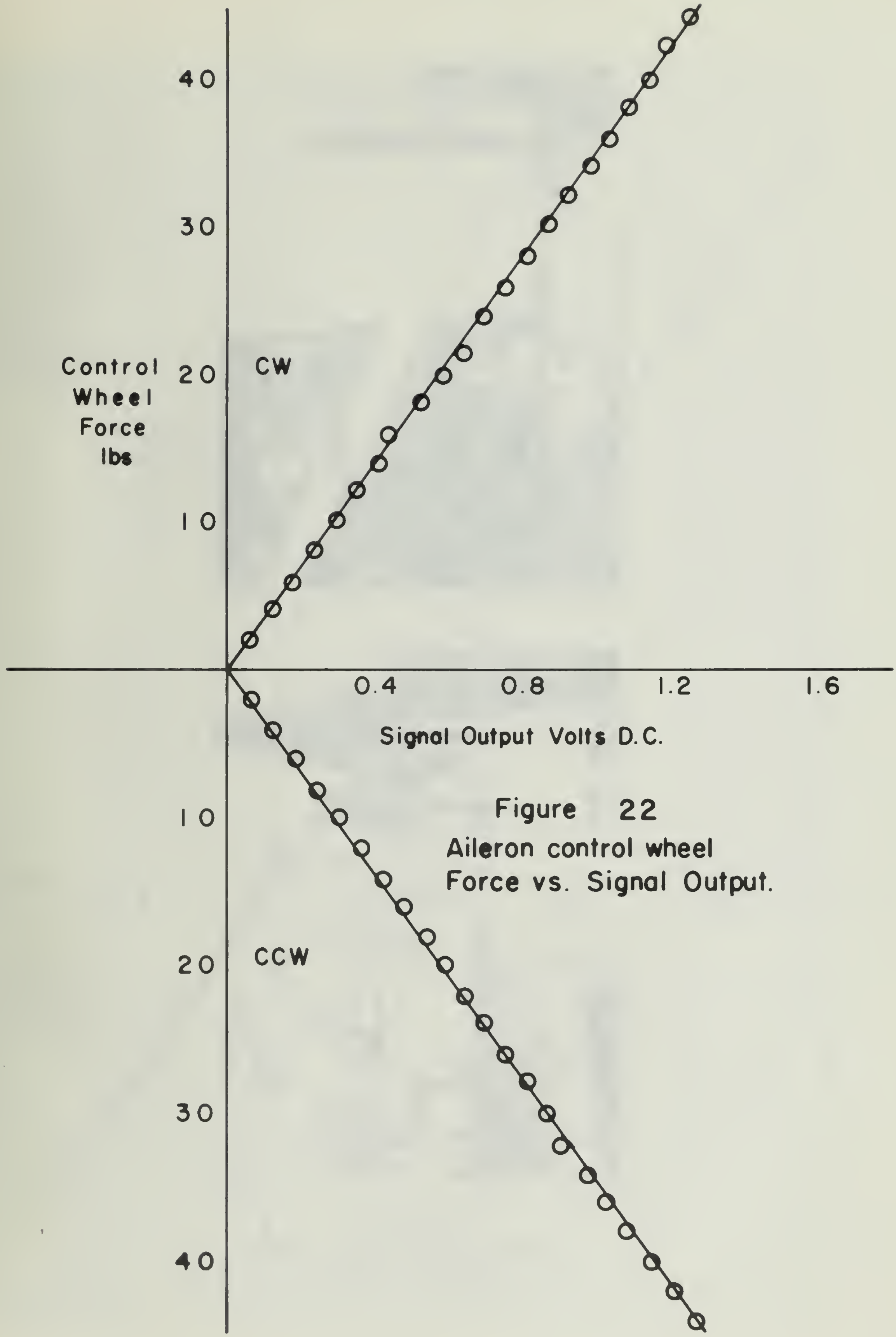


Figure 22
Aileron control wheel
Force vs. Signal Output.

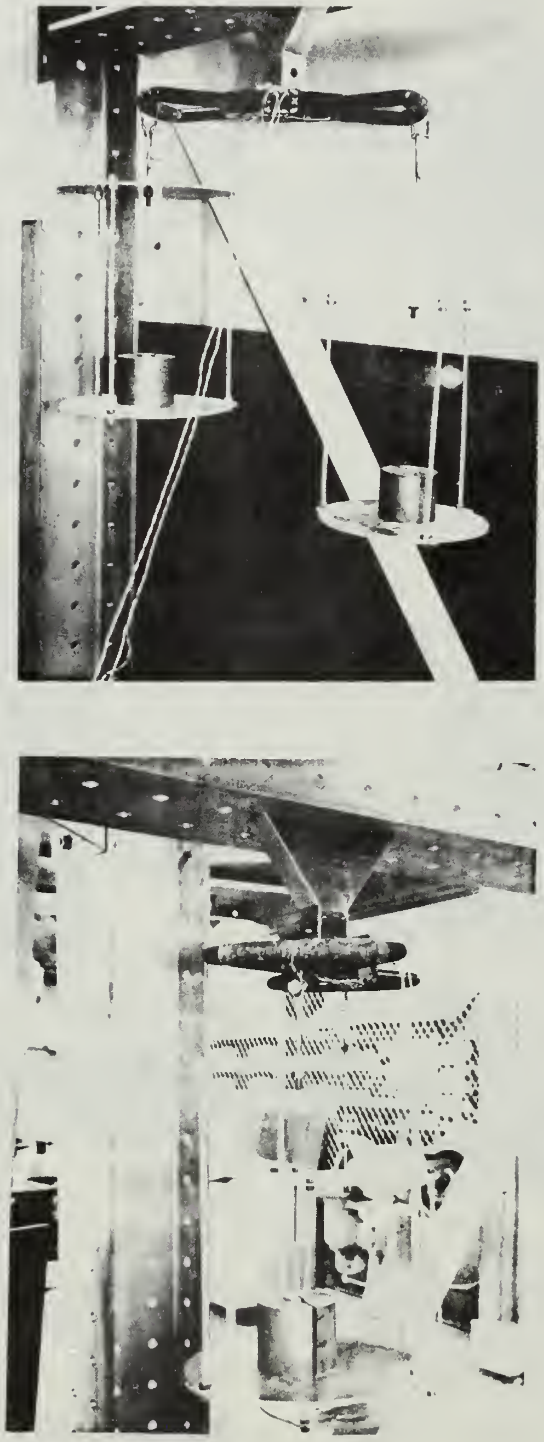


FIGURE 23
ELEVATOR CONTROL WHEEL FORCE
CALIBRATION SET-UP

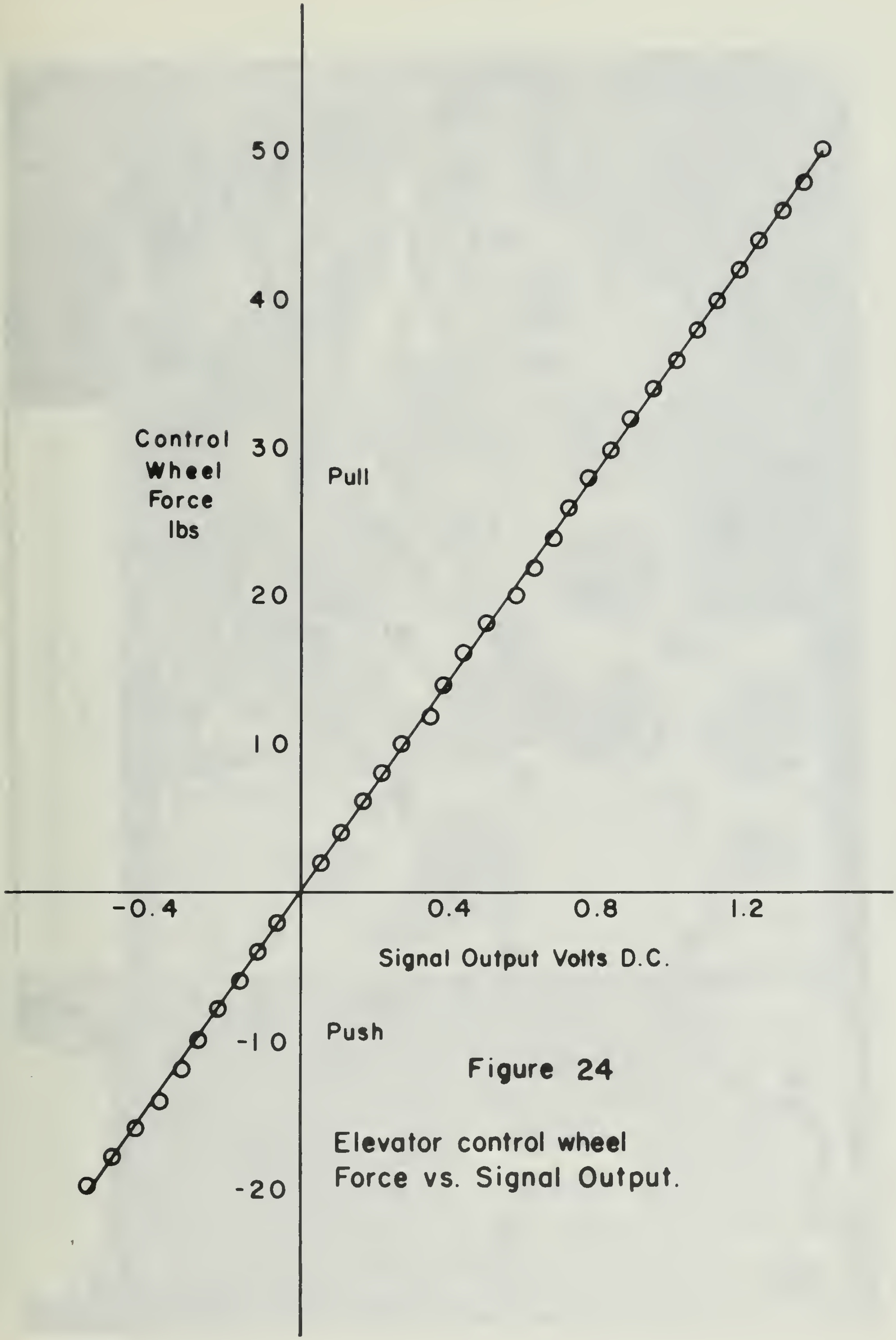


Figure 24

Elevator control wheel
Force vs. Signal Output.

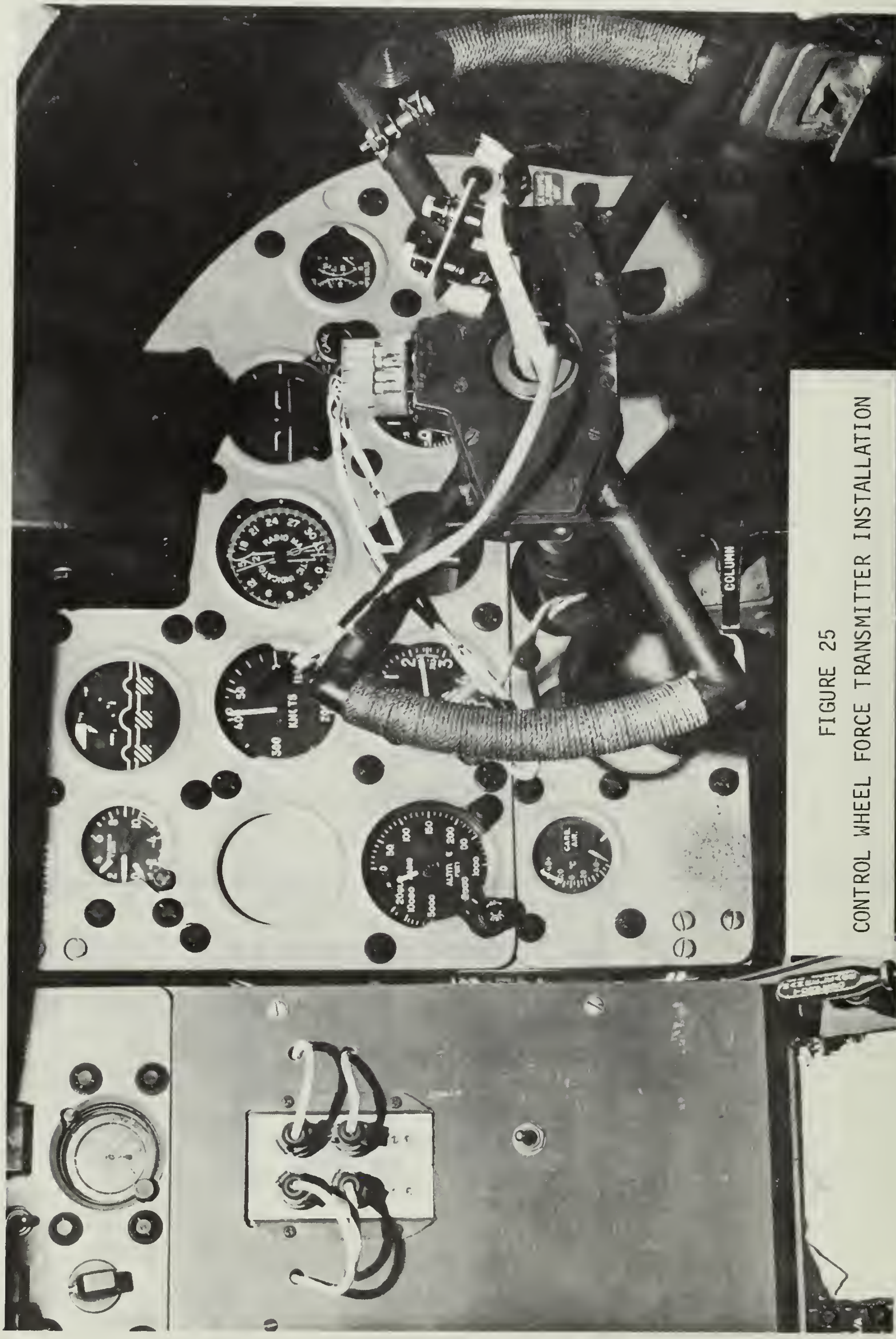


FIGURE 25
CONTROL WHEEL FORCE TRANSMITTER INSTALLATION

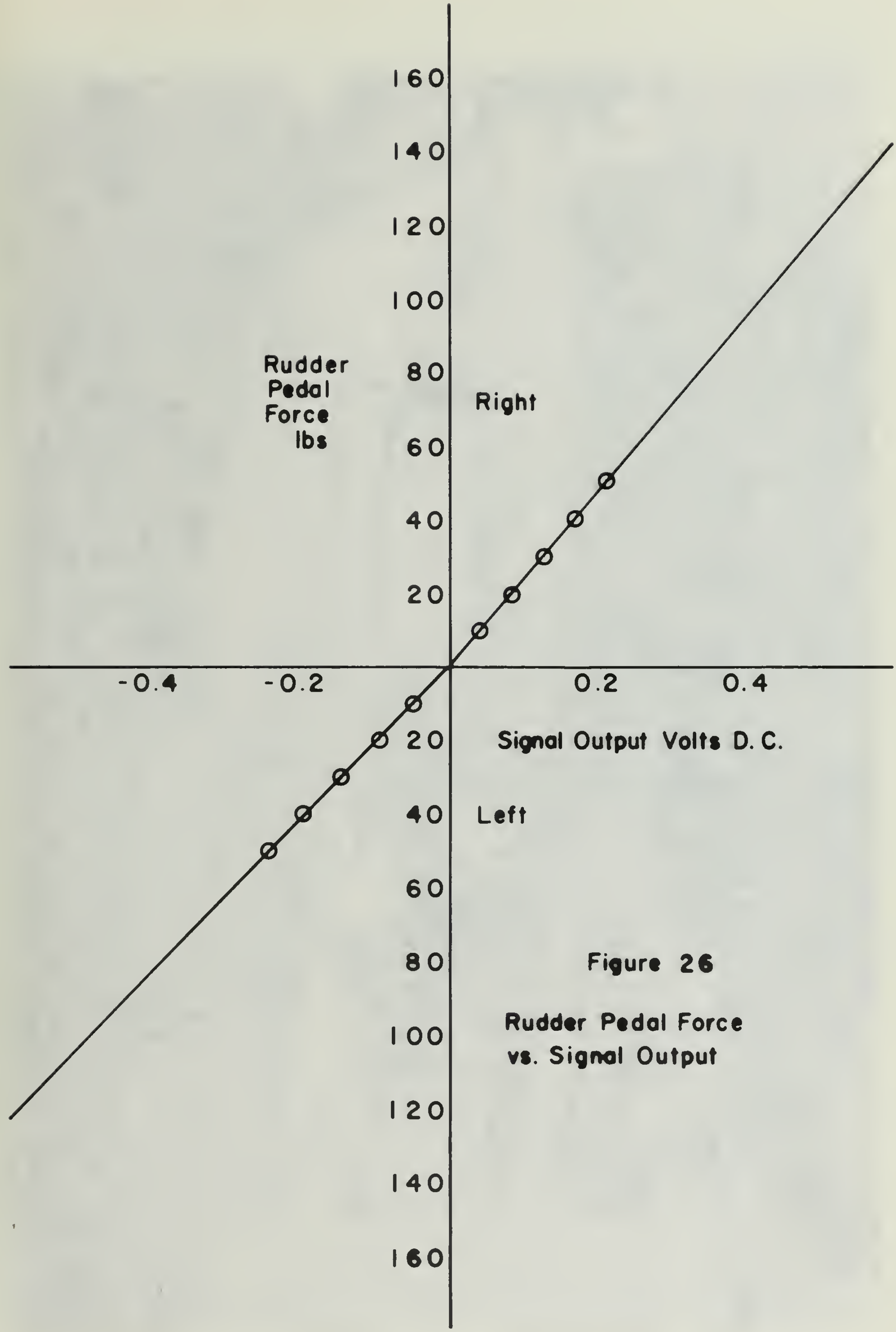


Figure 26

Rudder Pedal Force
vs. Signal Output

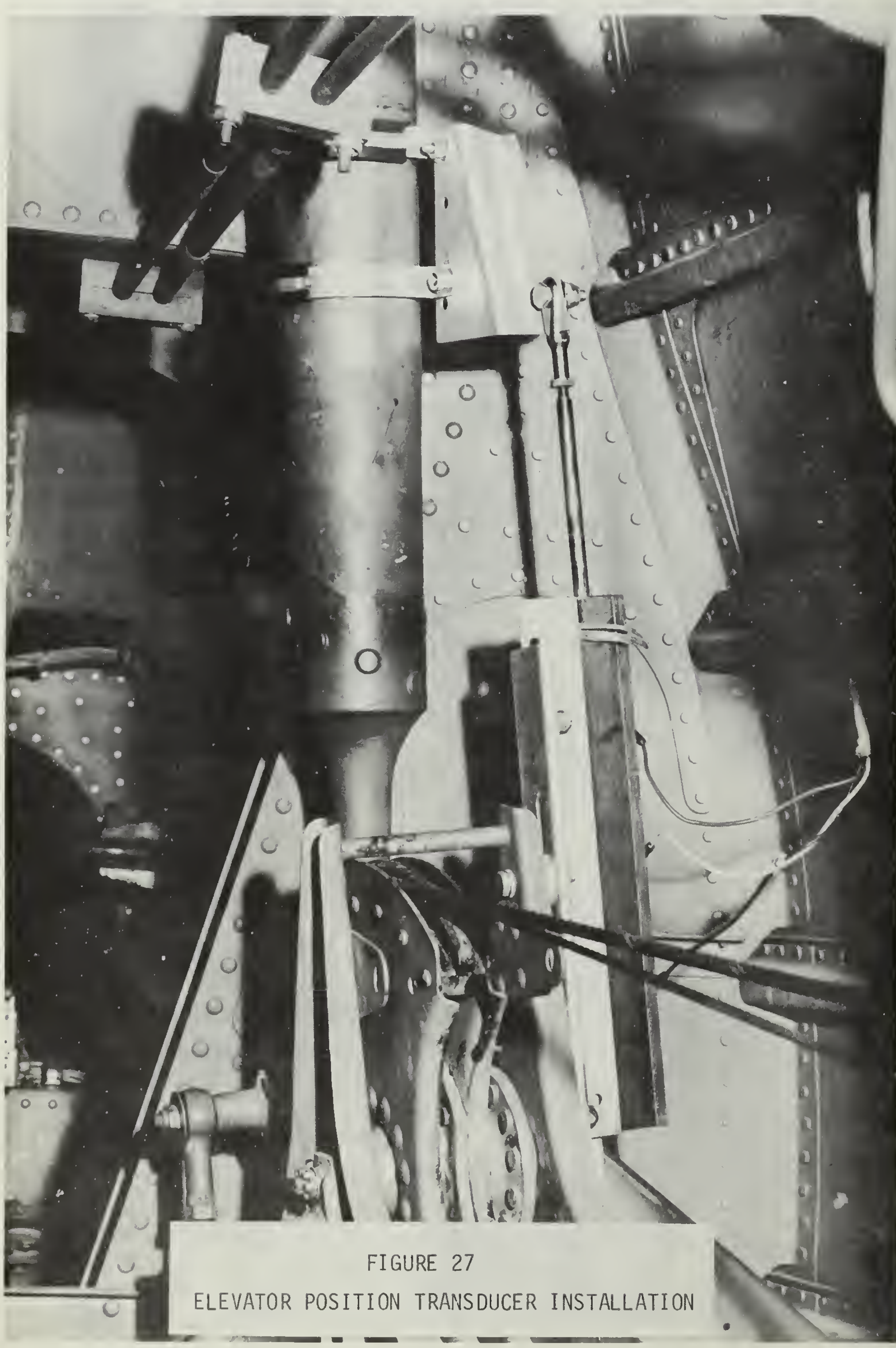


FIGURE 27

ELEVATOR POSITION TRANSDUCER INSTALLATION

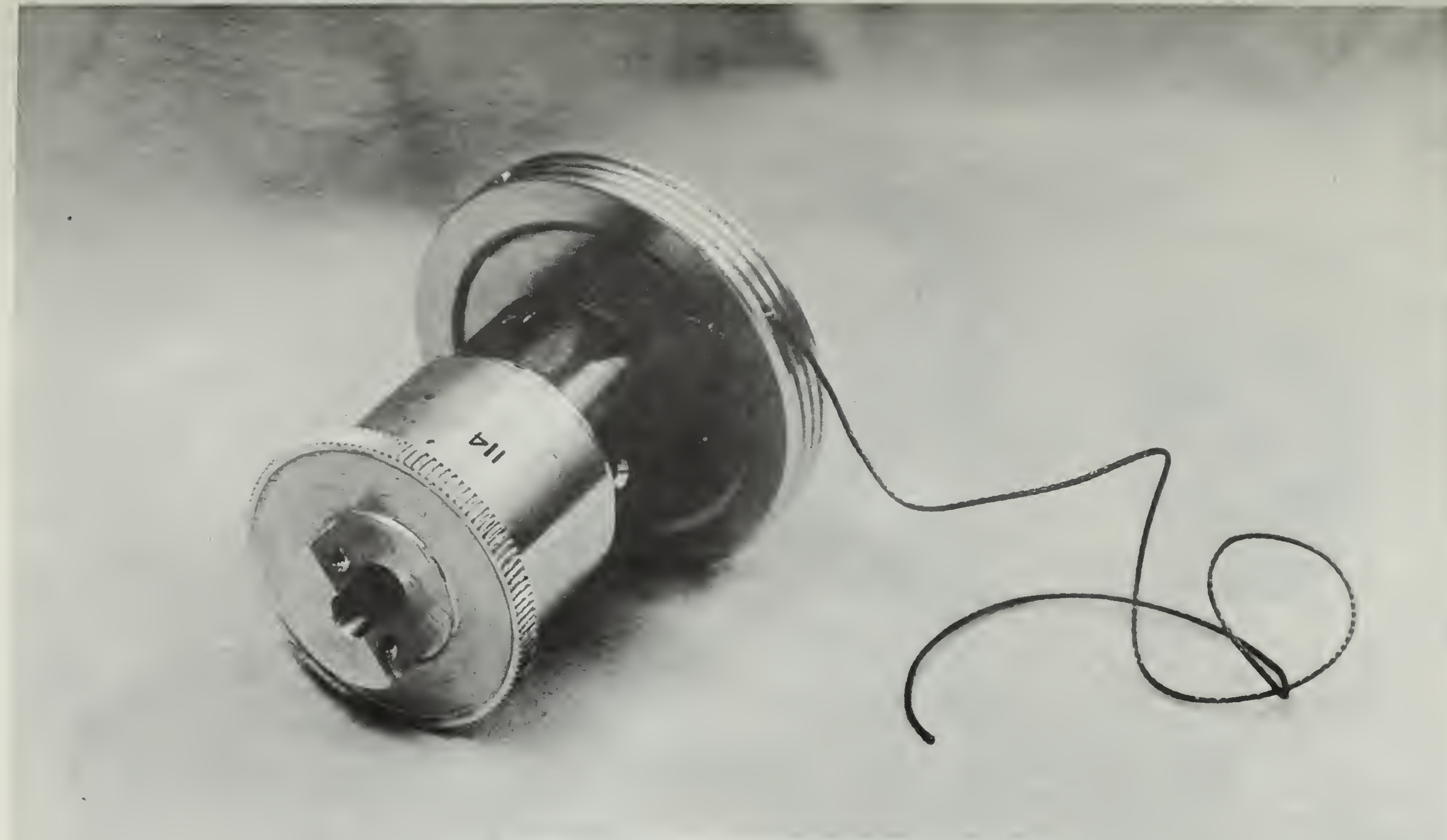


FIGURE 28
AILERON AND RUDDER POSITION TRANSDUCER

FIGURE 29

INTERNAL MECHANISMS OF THE AILERON
AND RUDDER POSITION TRANSDUCER

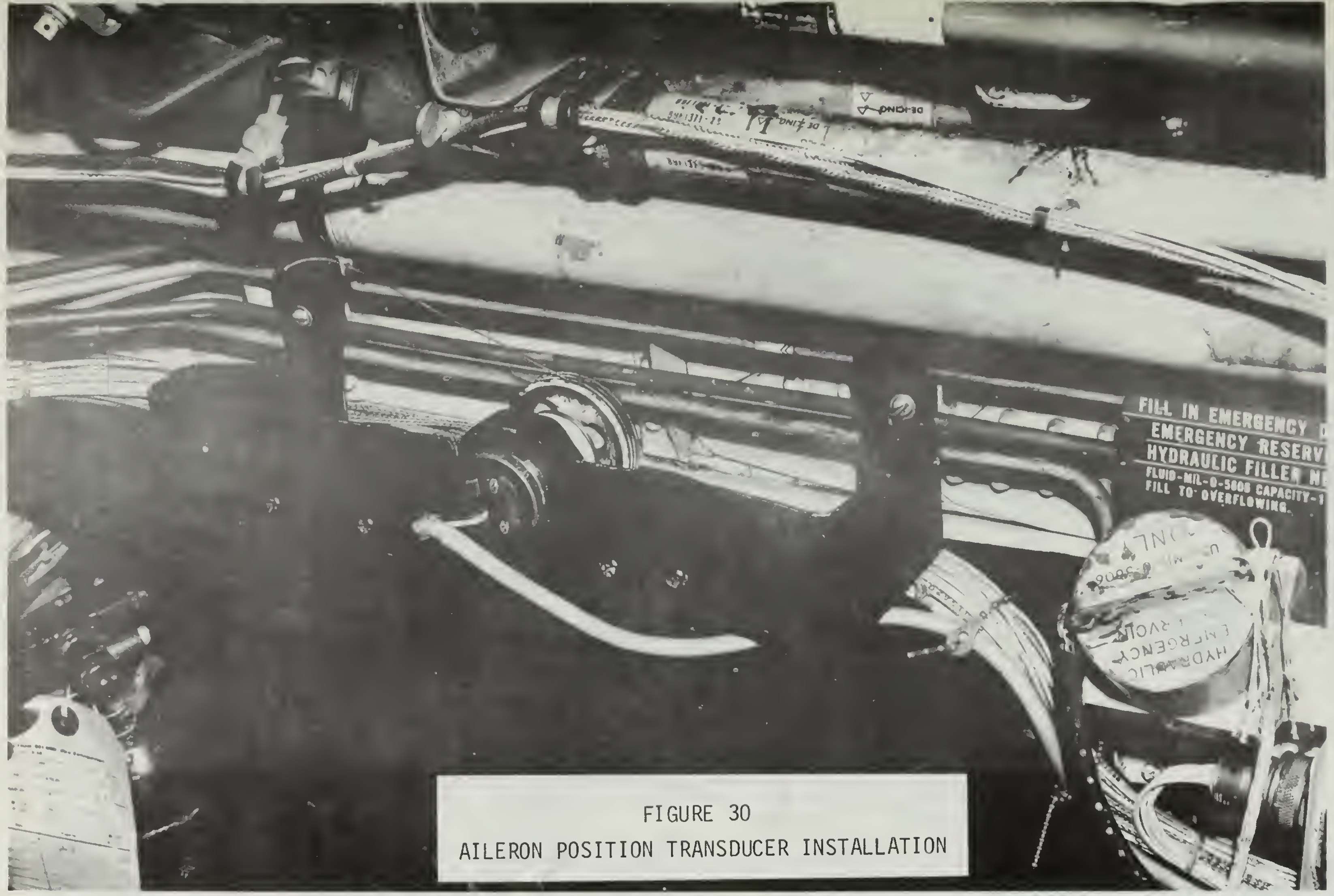


FIGURE 30
AILERON POSITION TRANSDUCER INSTALLATION

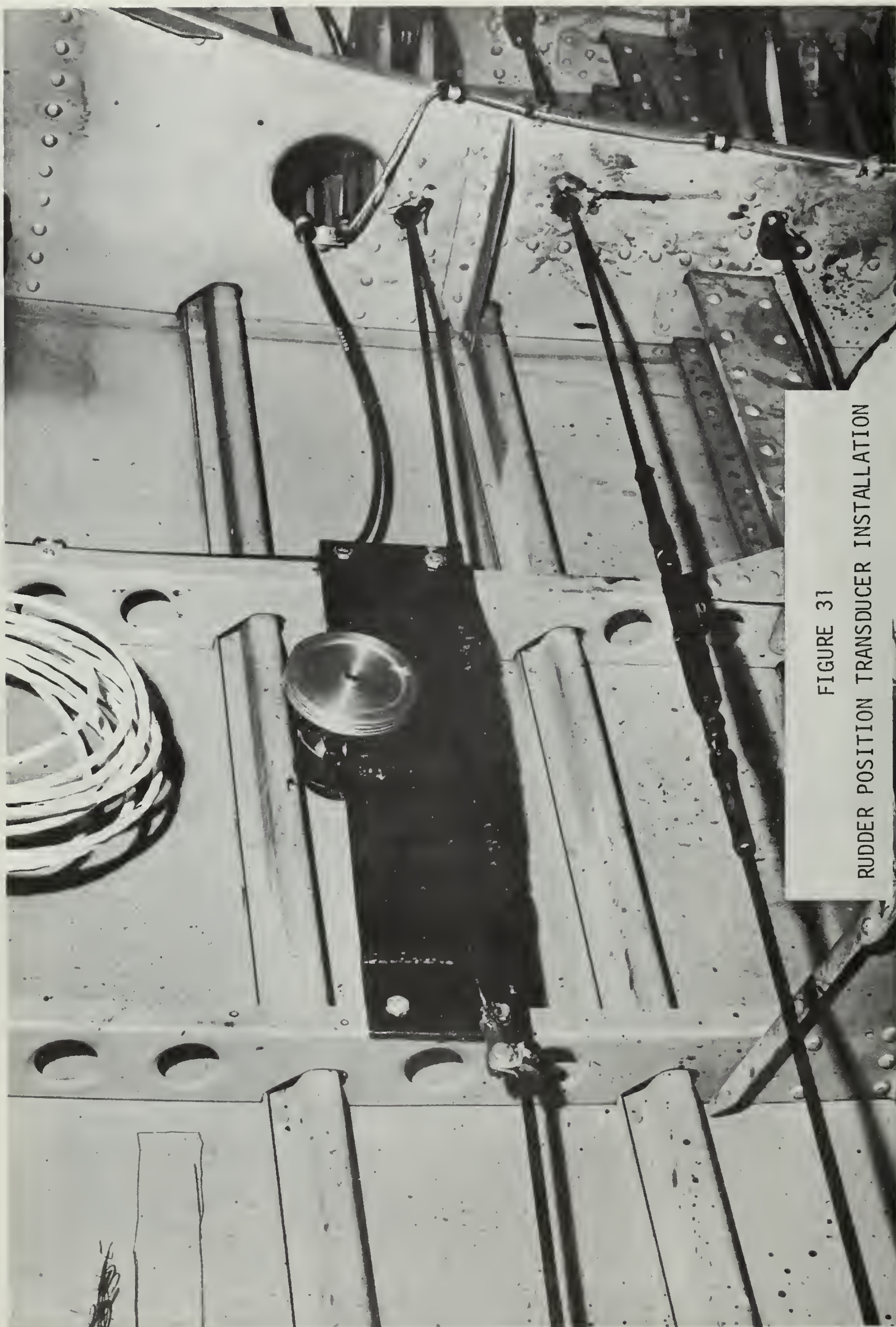


FIGURE 31
RUDDER POSITION TRANSDUCER INSTALLATION

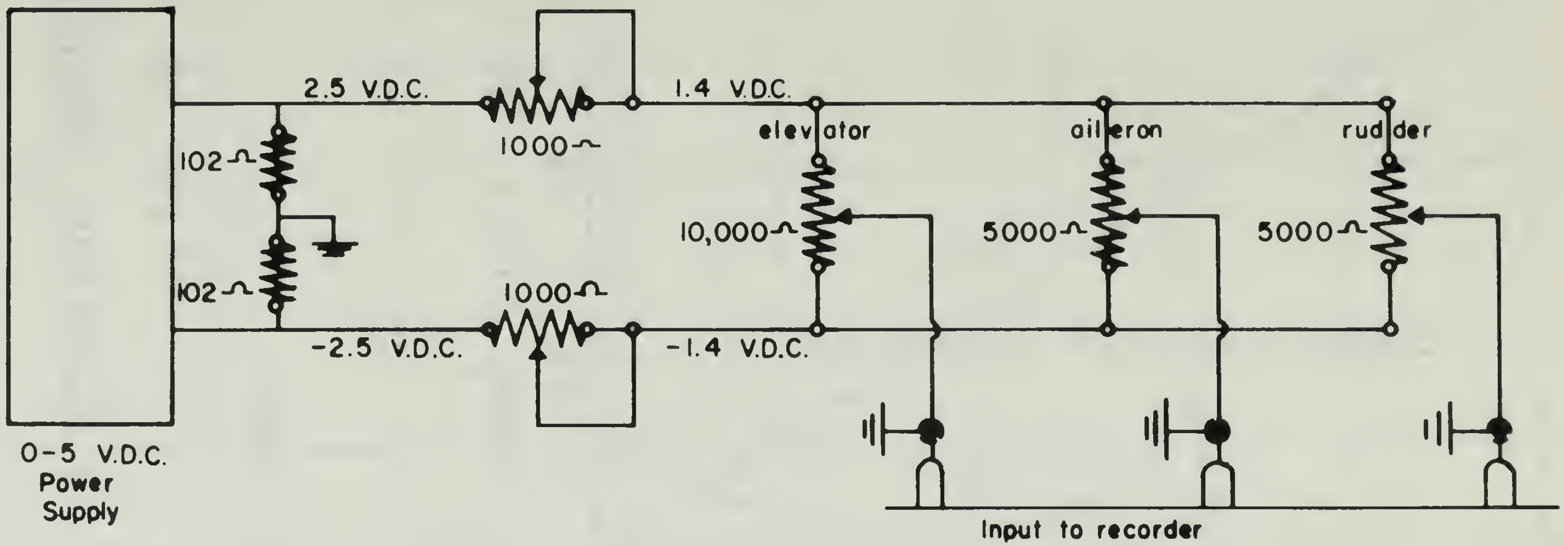


Figure 32

Control Surface Position Schematic.

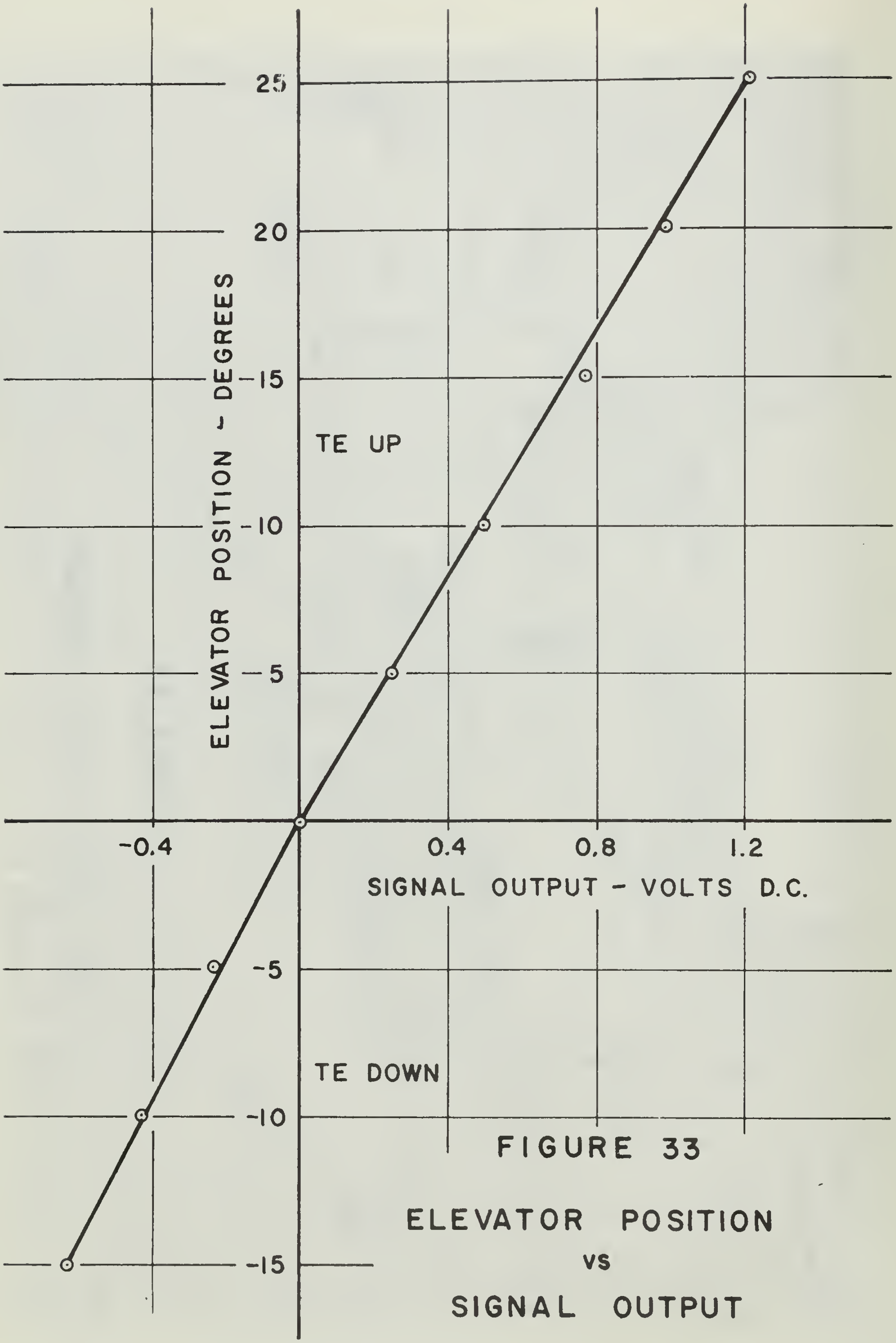


FIGURE 33

ELEVATOR POSITION
VS
SIGNAL OUTPUT

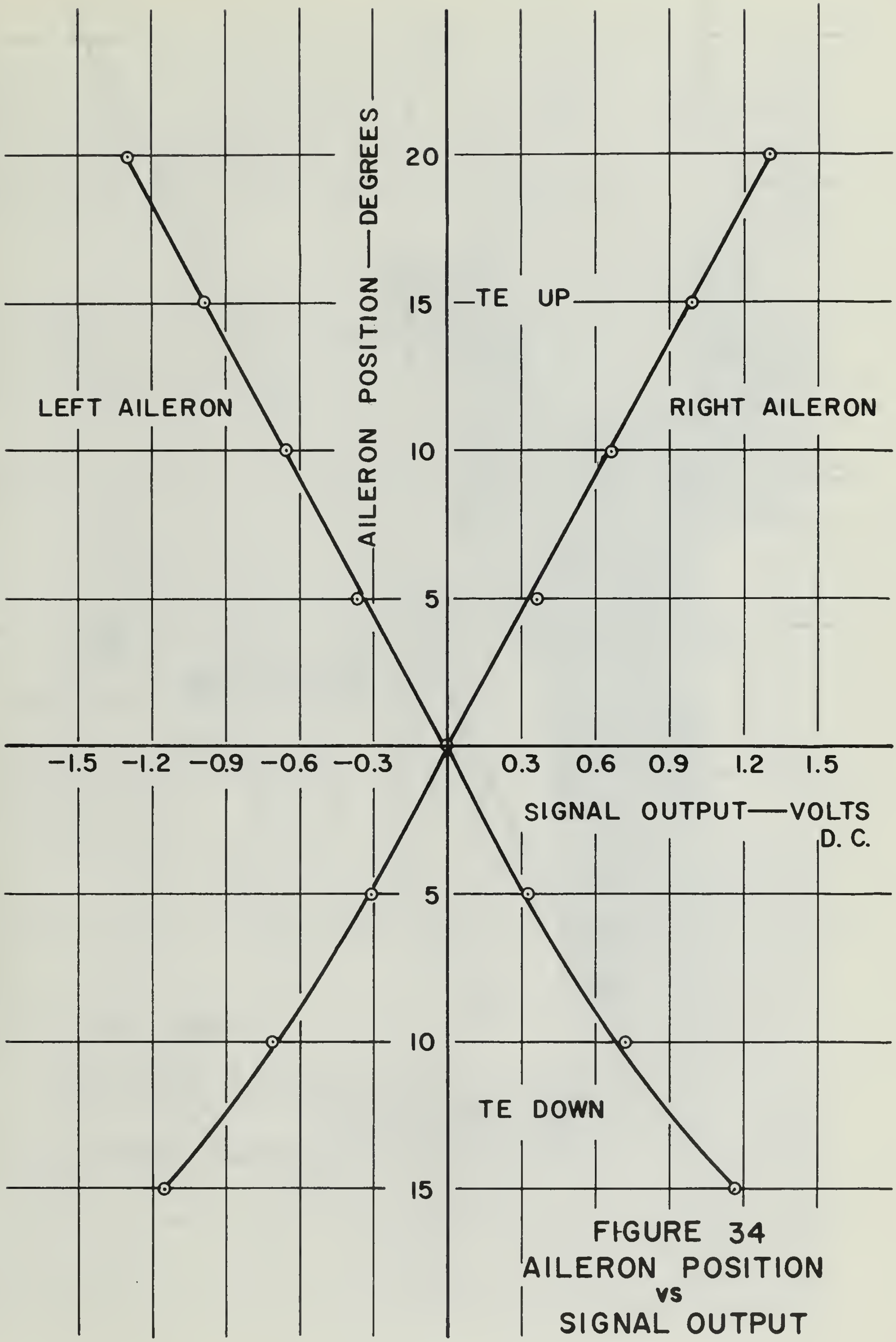
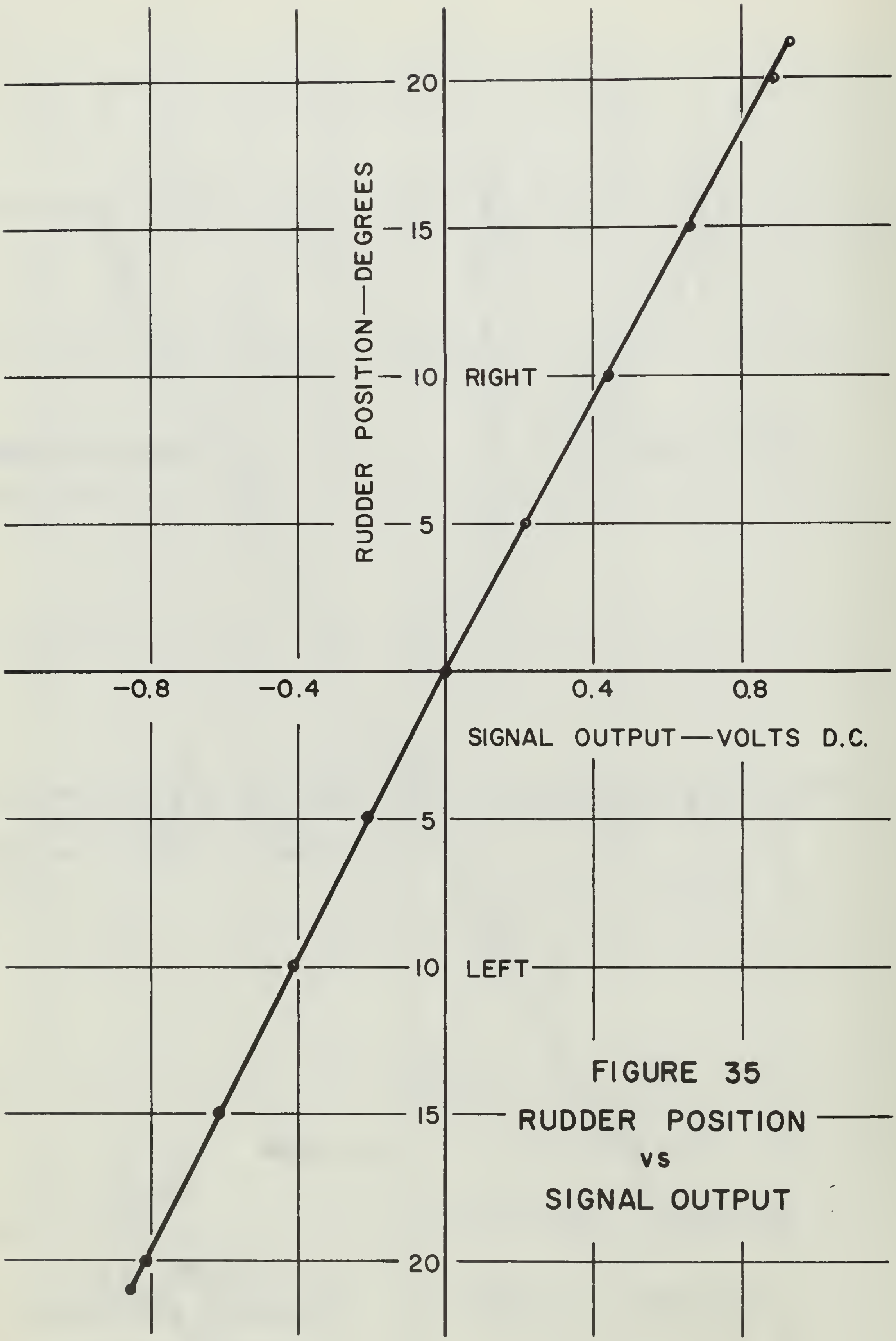


FIGURE 34
AILERON POSITION
VS
SIGNAL OUTPUT


RUDDER POSITION—DEGREES
20
15
10
5
5
10
15
20
RIGHT
LEFT
−0.8
−0.4
0.4
0.8
SIGNAL OUTPUT—VOLTS D.C.
FIGURE 35
RUDDER POSITION
VS
SIGNAL OUTPUT

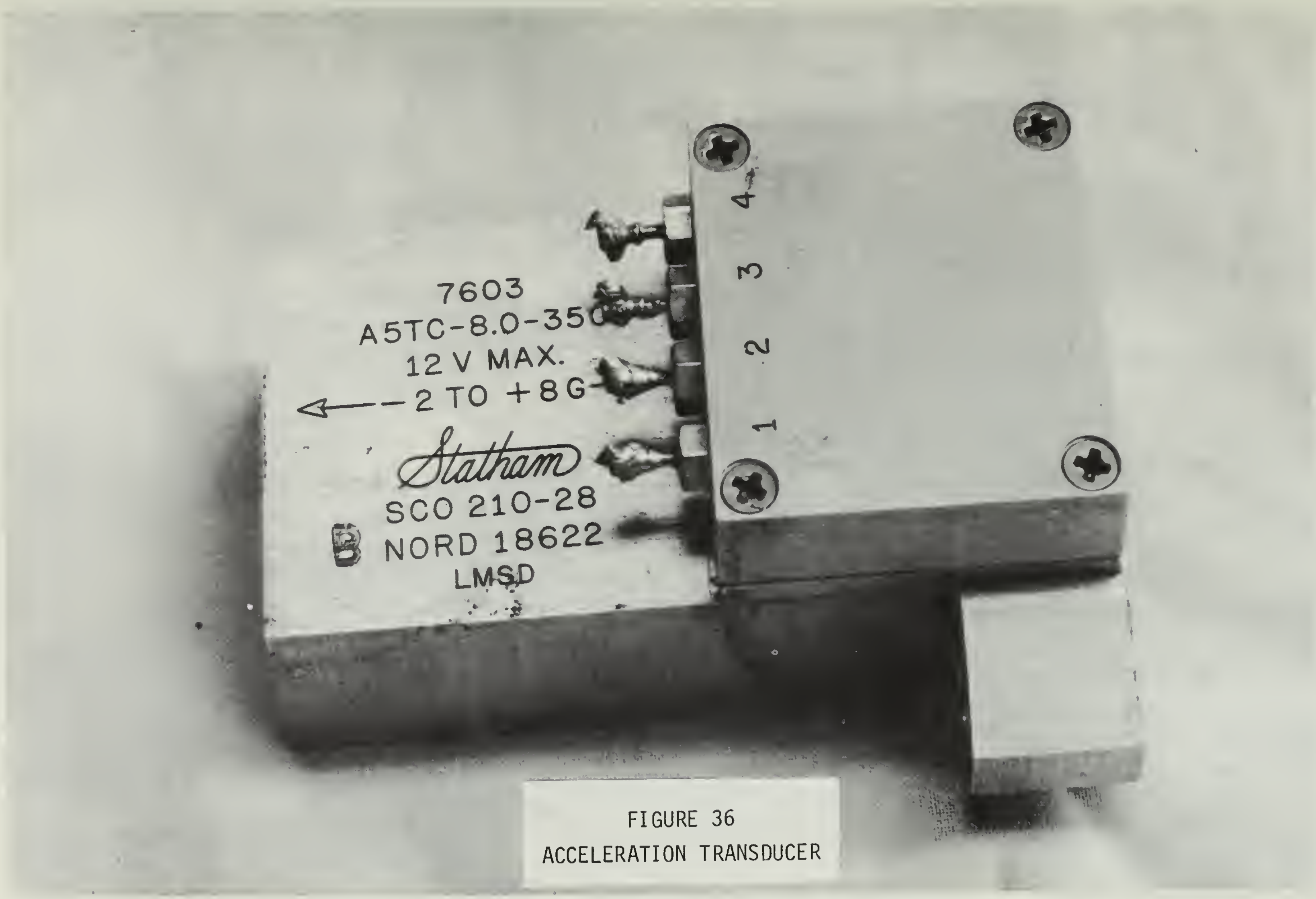


FIGURE 36
ACCELERATION TRANSDUCER

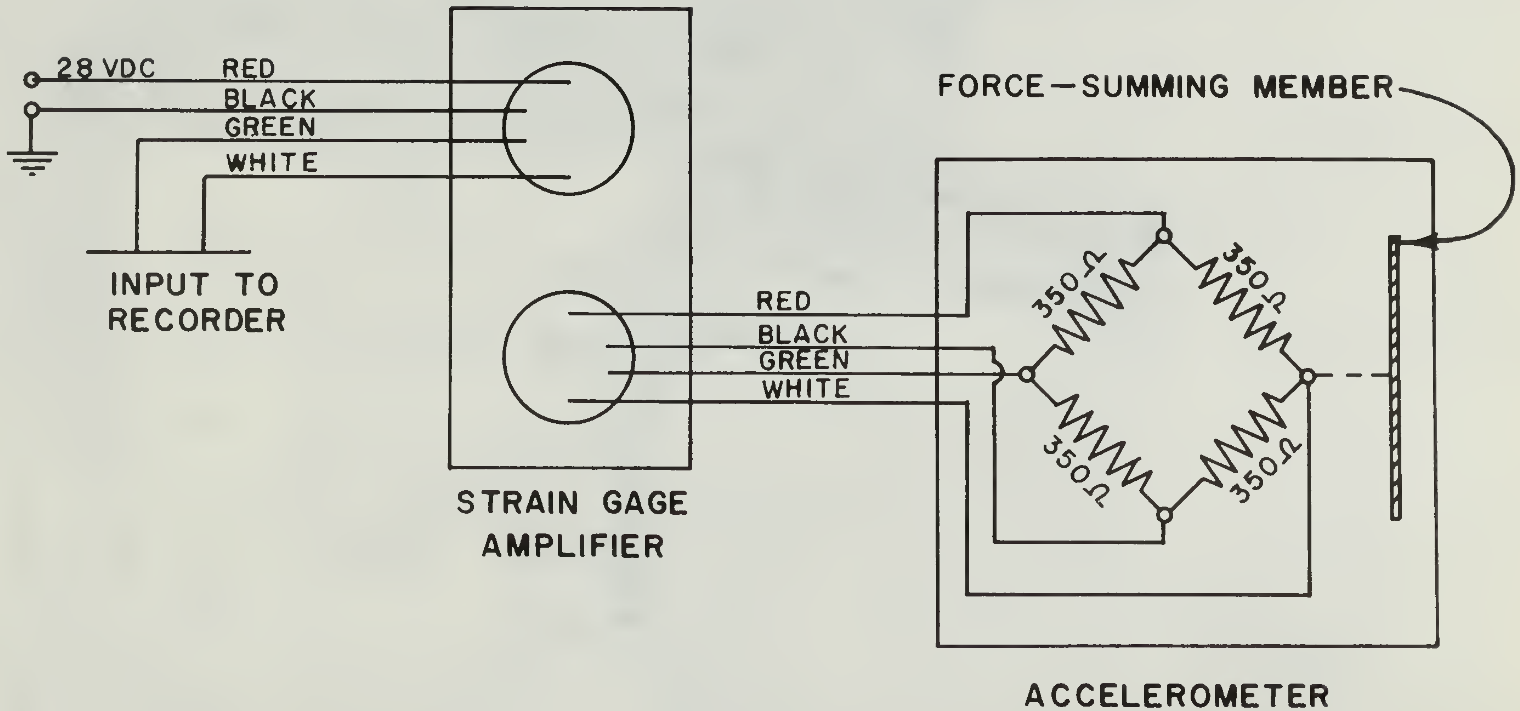


FIGURE 37

NORMAL ACCELERATION SCHEMATIC

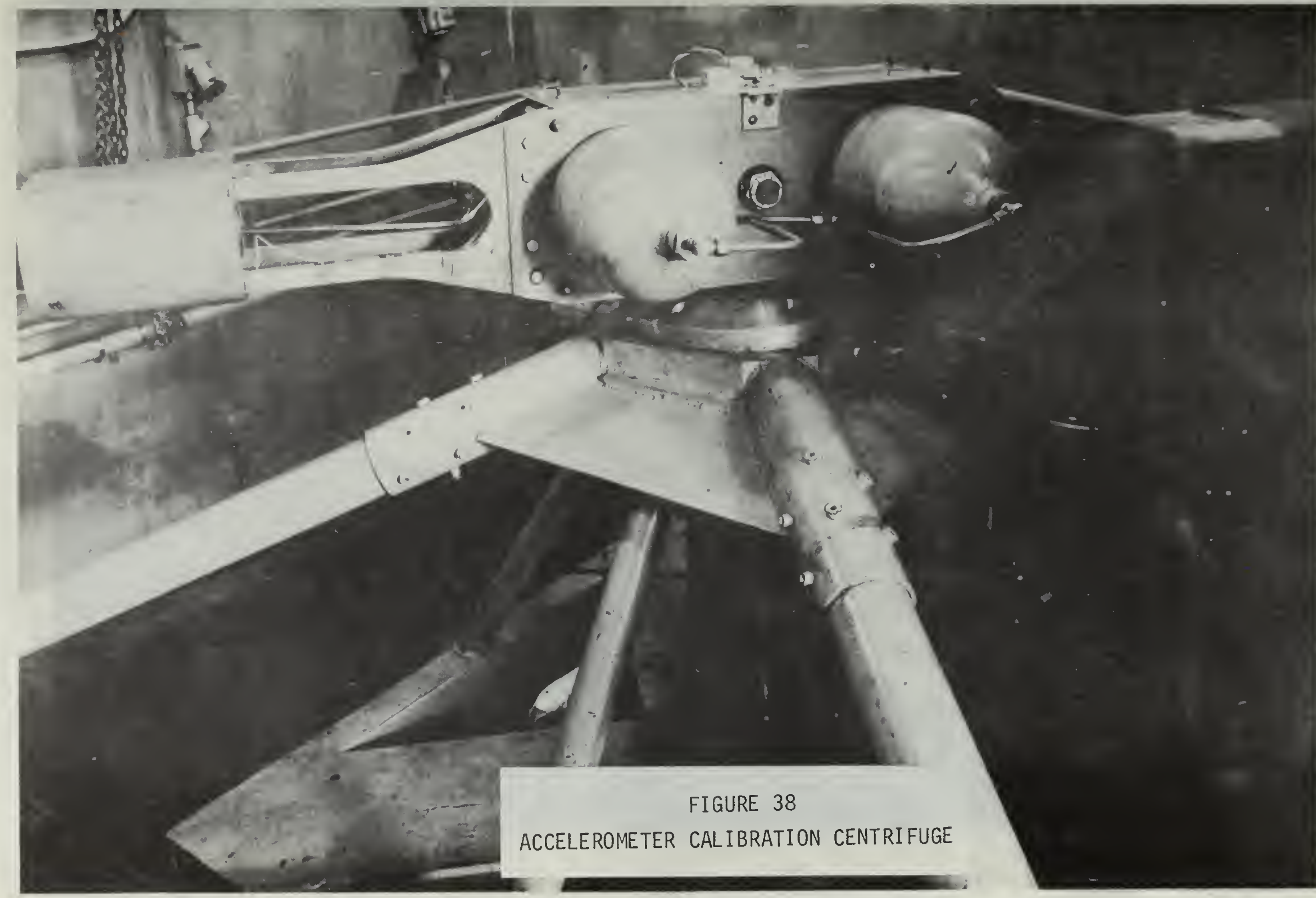


FIGURE 38
ACCELEROMETER CALIBRATION CENTRIFUGE

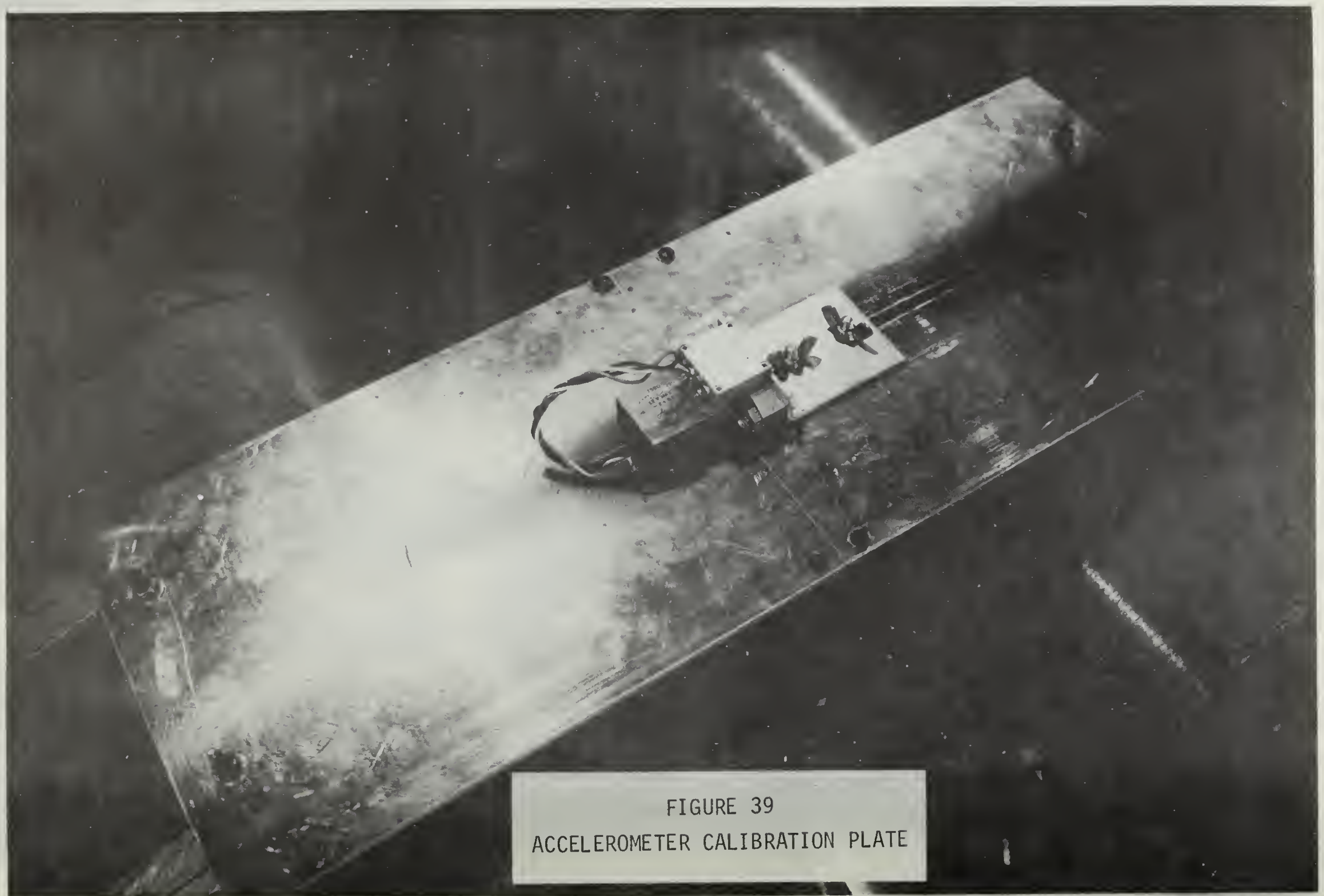


FIGURE 39
ACCELEROMETER CALIBRATION PLATE

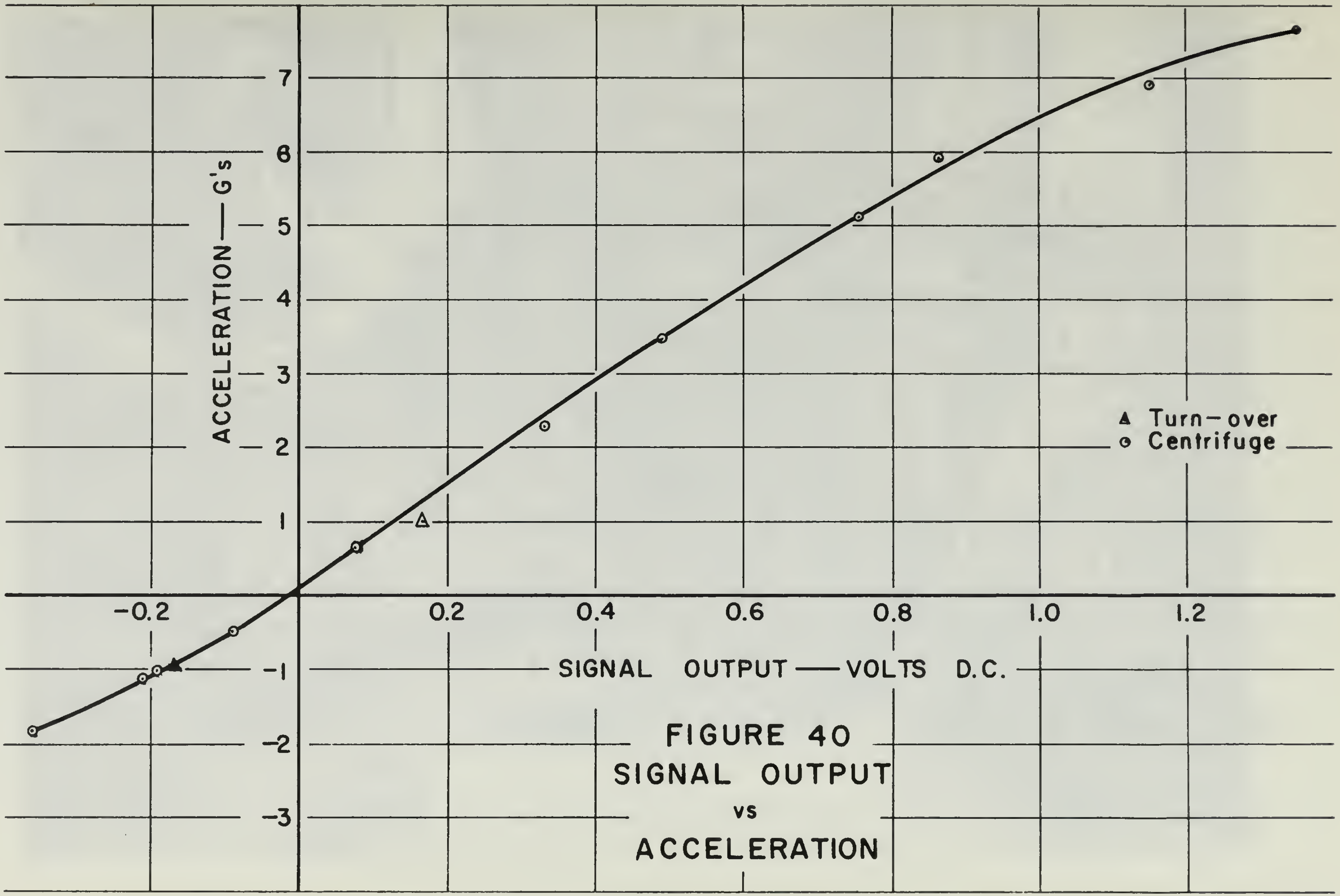


FIGURE 40
SIGNAL OUTPUT
vs
ACCELERATION

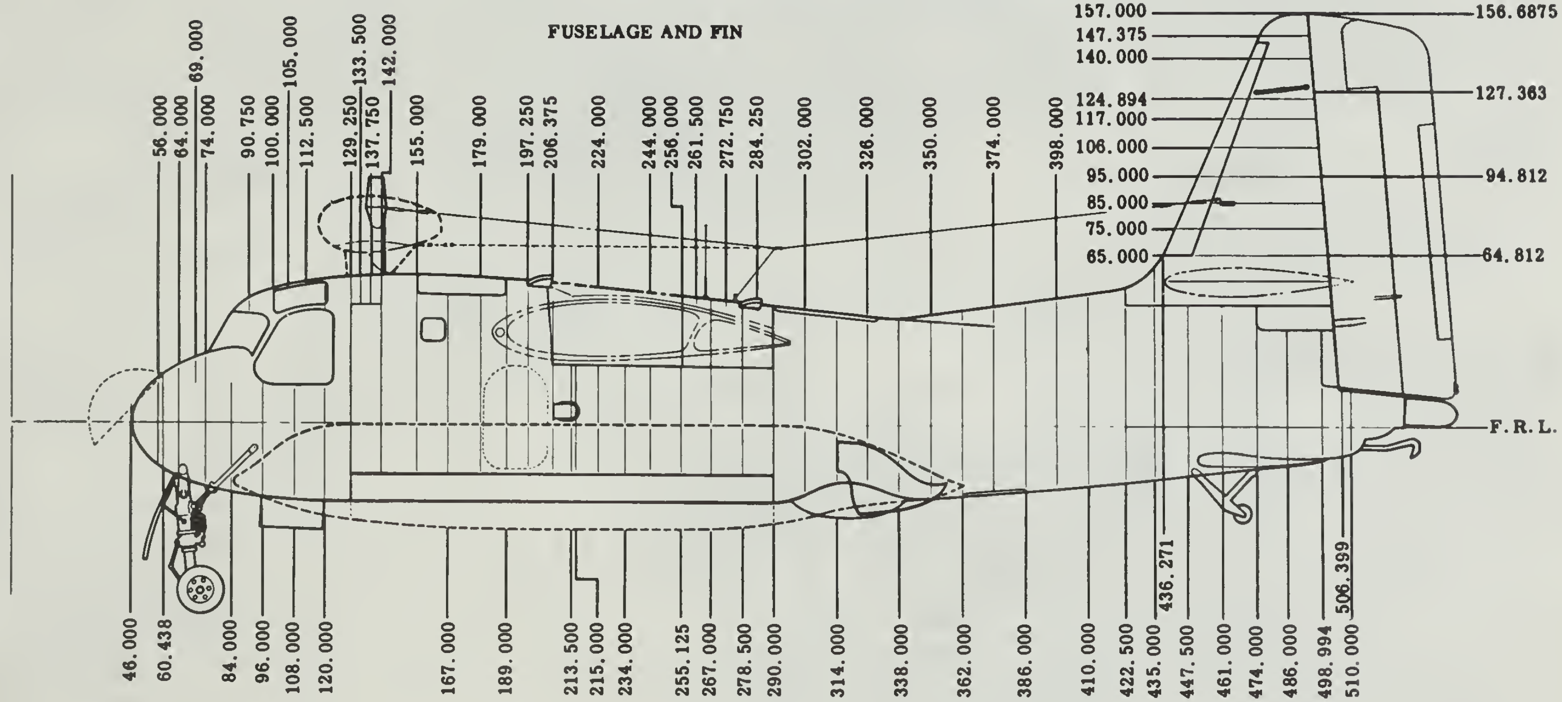


FIGURE 41

AIRCRAFT FUSELAGE FLIGHT STATION LOCATION

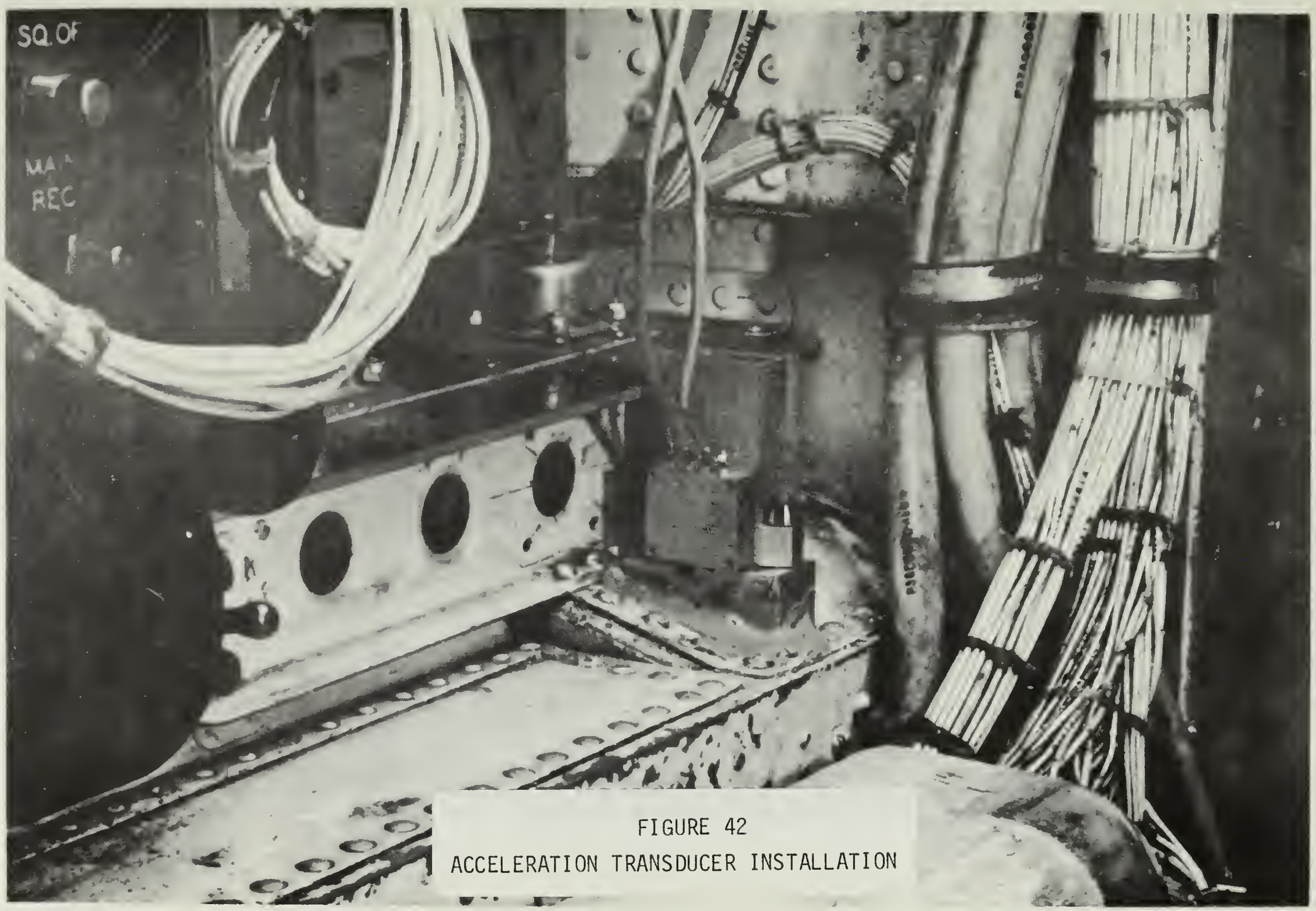


FIGURE 42
ACCELERATION TRANSDUCER INSTALLATION

APPENDIX B

ORIGINAL CALIBRATION DATA

TABLE II

AILERON CONTROL WHEEL FORCE CALIBRATION

Clockwise		Counterclockwise	
Force Pounds	Signal output Volts, dc (positive)	Force Pounds	Signal output Volts, dc (positive)
2	0.059	2	0.060
4	.121	4	.120
6	.179	6	.180
8	.231	8	.240
10	.296	10	.300
12	.353	12	.360
14	.413	14	.417
16	.462	16	.475
18	.524	18	.535
20	.581	20	.587
22	.638	22	.640
24	.688	24	.696
26	.750	26	.750
28	.805	28	.811
30	.863	30	.865
32	.917	32	.912
34	.978	34	.983
36	1.030	36	1.034
38	1.082	38	1.095
40	1.143	40	1.150
42	1.185	42	1.208
44	1.246	44	1.274
46	1.300	46	1.319
48	1.344	48	1.371
50	1.394	50	1.430

TABLE III

ELEVATOR CONTROL WHEEL FORCE CALIBRATION

Push		Pull	
Force Pounds	Signal output Volts, dc (negative)	Force Pounds	Signal output Volts, dc (positive)
2	0.059	2	0.053
4	.118	4	.105
6	.172	6	.164
8	.226	8	.216
10	.274	10	.273
12	.323	12	.327
14	.384	14	.385
16	.448	16	.437
18	.507	18	.504
20	0.574	20	.571
		22	.629
		24	.682
		26	.723
		28	.778
		30	.832
		32	.889
		34	.951
		36	1.009
		38	1.064
		40	1.123
		42	1.180
		44	1.235
		46	1.298
		48	1.356
		50	1.409

TABLE IV

RUDDER PEDAL FORCE CALIBRATION

Force Pounds	Signal Output, Volts, dc Right Pedal	Left Pedal
0	0.000	0.000
10	.041	-0.048
20	.085	.096
30	.129	.147
40	.170	.199
50	0.212	-0.246

TABLE V

ELEVATOR POSITION CALIBRATION

Trailing Edge Up		Trailing Edge Down	
Position Degrees	Signal output Volts, dc (positive)	Position Degrees	Signal output Volts, dc (negative)
0	0.000	0	0.000
5	.248	5	.238
10	.498	10	.435
15	.776	15	0.630
20	.990		
25	1.219		

TABLE VI

RIGHT AILERON POSITION CALIBRATION

Trailing Edge Up		Trailing Edge Down	
Position Degrees	Signal output Volts, dc (positive)	Position Degrees	Signal output Volts, dc (negative)
0	0.000	0	0.000
5	.365	5	.314
10	.658	10	.712
15	.988	15	1.156
20	1.298		

TABLE VII

RUDDER POSITION CALIBRATION

Trailing Edge Right		Trailing Edge Left	
Position Degrees	Signal output Volts, dc (positive)	Position Degrees	Signal output Volts, dc (negative)
0	0.000	0	0.000
5	.215	5	.211
10	.440	10	.416
15	.663	15	.617
20	.886	20	.815
21.5	0.926	21.5	0.859

TABLE VIII

NORMAL ACCELERATION CALIBRATION

Centrifuge speed RPM	Accelerations* g's	Signal output Volts, dc
000	0.000	0.000
208	.614	.074
400	2.272	.330
494	3.465	.490
600	5.112	.755
644	5.889	.865
696	6.879	1.150
734	7.650	1.350
000	0.000	0.000
190	-0.513	-0.091
270	1.035	.195
280	1.113	.213
360	-1.840	-0.363

* $a = \frac{4\pi^2 N^2 r}{g(12)(3600)} = 2.840 \times 10^{-5} N^2 r$

N = centrifuge speed in RPM

r = ± 0.5 in

a = acceleration in g's

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Security Classification

DOCUMENT CONTROL DATA - R & D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

1. ORIGINATING ACTIVITY *(Corporate author)* Naval Postgraduate School Monterey, California 93940	2a. REPORT SECURITY CLASSIFICATION Unclassified 2b. GROUP

3. REPORT TITLE

THE INSTRUMENTATION OF THE S-2 AIRCRAFT FOR STABILITY AND CONTROL FLIGHT TESTING

4. DESCRIPTIVE NOTES *(Type of report and inclusive dates)*
Master's Thesis (April 1969)

5. AUTHOR(S) *(First name, middle initial, last name)*

Thomas John Duncan

6. REPORT DATE	7a. TOTAL NO. OF PAGES	7b. NO. OF REFS
April 1969	94	10

8a. CONTRACT OR GRANT NO. b. PROJECT NO. c. d.	9a. ORIGINATOR'S REPORT NUMBER(S) 9b. OTHER REPORT NO(S) *(Any other numbers that may be assigned this report)*

10. DISTRIBUTION STATEMENT

Distribution of this document is unlimited.

11. SUPPLEMENTARY NOTES	12. SPONSORING MILITARY ACTIVITY
	Naval Postgraduate School Monterey, California 93940

13. ABSTRACT

A Navy US-2A aircraft was instrumented for use in stability and control flight testing. Various methods of recording and sensing the aerodynamic parameters necessary to evaluate stability and control flight testing were investigated. An Ampex Series 800 Magnetic Tape Recorder, obtained on a loan basis, was calibrated and installed to provide a means of recording airborne data. Using available equipment, sensing devices were installed in the aircraft and calibrated to measure control forces, control surface position and normal acceleration. The completed installation provides the means for sensing and recording those aerodynamic parameters most difficult to measure without electronic aids. It also allows for the incorporation of additional sensing and recording devices should they become available.

DD FORM 1 NOV 65 1473 (PAGE 1)

S/N 0101-807-6811

Unclassified

Security Classification

A-31408

14 KEY WORDS	LINK A		LINK B		LINK C	
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Flight Testing						
Control and Stability						
Data Acquisition System						
Control Forces						

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